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*Technical Memorandum 33-583*

*Volume II*

*Solar Electric Propulsion System Integration  
Technology (SEPSIT) Final Report*

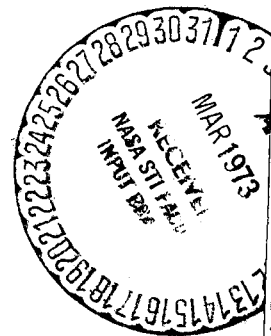
*Encke Rendezvous Mission and Space Vehicle  
Functional Description*

*J. A. Gardner*

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PASADENA, CALIFORNIA**

November 15, 1972



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National Aeronautics and Space Administration

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## CONTENTS

## SECTION

I.	INTRODUCTION . . . . .	I-1
II.	FUNCTIONAL DESCRIPTION . . . . .	II-A-1
A.	SCOPE . . . . .	II-A-1
B.	REQUIREMENTS AND CONSTRAINTS . . . . .	II-B-1
C.	ENCKE '80 RENDEZVOUS MISSION DESCRIPTION . . . . .	II-C-1
	1. Trajectory Characteristics . . . . .	II-C-1
	2. Sequence of Events . . . . .	II-C-23
D.	SPACE VEHICLE SYSTEM DESCRIPTION . . . . .	II-D-1
	1. Configuration . . . . .	II-D-1
	2. Equipment List, Weights and Spacecraft Mass Properties . . . . .	II-D-10
	3. Space Vehicle System Functional Block Diagram . . . . .	II-D-13
	4. Data Handling and Commands . . . . .	II-D-17
	5. Power Management . . . . .	II-D-20
	6. Electronic Packaging . . . . .	II-D-20
E.	SPACECRAFT SUBSYSTEM DESCRIPTIONS . . . . .	II-E-1
	1. Power Subsystem . . . . .	II-E-1
	2. Attitude Control . . . . .	II-E-4
	3. Telecommunications Subsystem . . . . .	II-E-10
	4. Computer Command Subsystem . . . . .	II-E-15
	5. Flight Data Subsystem . . . . .	II-E-23
	6. Data Storage Subsystem . . . . .	II-E-25
	7. Structure Subsystem (STRU) . . . . .	II-E-30
	8. Cabling Subsystem (CABL) . . . . .	II-E-32
	9. Mechanical Devices Subsystem (DEV) . . . . .	II-E-34
	10. Thermal Control Subsystem (TC) . . . . .	II-E-35
	11. Science Instruments . . . . .	II-E-37
F.	SEP MODULE SUBSYSTEM DESCRIPTIONS . . . . .	II-F-1
	1. Thrust Subsystem (TSS) . . . . .	II-F-1
	2. SEP Module Power Subsystem (MPSS) . . . . .	II-F-7

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## CONTENTS (Contd)

## SECTION

3.	Master Flight Data Subsystem (MFDS) . . . . .	II-F-11
4.	SEP Module Structure Subsystem (MSTRU) . . . . .	II-F-14
5.	Attitude Control Subsystem (TVC) . . . . .	II-F-17
6.	SEP Module Cabling Subsystem (MCABL) . . . . .	II-F-20
7.	SEP Module Mechanical Devices Sub- system (MDEV) . . . . .	II-F-24

## APPENDICES

APPENDIX A.	THRUST SUBSYSTEM DETAILED FUNCTIONAL DESCRIPTION . . . . .	A-1
APPENDIX B.	DESIGN REQUIREMENT, THRUST SUBSYSTEM FLIGHT ELECTRONIC PACKAGING . . . . .	B-1
APPENDIX C.	FLIGHT EQUIPMENT, THRUST SUBSYSTEM CABLING AND CABLING HARNESSES . . . . .	C-1

## TABLES

II-D-1.	Equipment List and Mass Allocations . . . . .	II-D-11
II-D-2.	CG and Inertial Properties . . . . .	II-D-12
II-D-3.	Power Utilization Summary . . . . .	II-D-21
II-E-1.	DSS Functional Characteristics . . . . .	II-E-27
II-E-2.	Science Payload Complement for an Encke Rendezvous Mission . . . . .	II-E-38

## FIGURES

I-1.	Space Vehicle . . . . .	I-2
II-C-1.	Normalized Solar Power Profile . . . . .	II-C-3
II-C-2.	Flight Time and Performance Trade Data for Direct Rendezvous Trajectories to Encke (1980 Perihelion). . . . .	II-C-3
II-C-3.	Encke Rendezvous (1980) Nominal Trajectory Ecliptic Projection . . . . .	II-C-4
II-C-4.	1980 Encke Rendezvous Mission, Thruster Subsystem Power Profile for Nominal Trajectory . . . . .	II-C-5
II-C-5.	Body-Fixed Coordinate System for Solar Electric Vehicle . . . . .	II-C-7

## CONTENTS (Contd)

## FIGURES

II-C-6.	Sun-Vehicle-Thrust Beam Angle . . . . .	II-C-8
II-C-7.	Thrust Plane Roll Angle . . . . .	II-C-8
II-C-8.	Thrust Beam Orientation at Thrust Initiation . . . . .	II-C-9
II-C-9.	Propellant Consumption for Nominal Trajectory . . . . .	II-C-10
II-C-10.	Sun-Vehicle Distance . . . . .	II-C-11
II-C-11.	Sun-Vehicle-Encke Angle . . . . .	II-C-11
II-C-12.	Earth-Vehicle Range . . . . .	II-C-12
II-C-13.	Earth-Vehicle Range Rate . . . . .	II-C-12
II-C-14.	Earth-Co-Elevation Angle . . . . .	II-C-13
II-C-15.	Earth Azimuth Angle . . . . .	II-C-13
II-C-16.	Vehicle Right Ascension from Earth . . . . .	II-C-14
II-C-17.	Vehicle Declination from Earth . . . . .	II-C-14
II-C-18.	Encke Right Ascension and Declination from Vehicle . . . . .	II-C-15
II-C-19.	Encke-Vehicle Range . . . . .	II-C-15
II-C-20.	Encke-Vehicle Range Rate . . . . .	II-C-16
II-C-21.	Encke Co-Elevation Angle . . . . .	II-C-16
II-C-22.	Encke Azimuth Angle . . . . .	II-C-17
II-C-23.	Encke - Vehicle - Thrust Beam Angle . . . . .	II-C-17
II-C-24.	Canopus Co-Elevation Angle . . . . .	II-C-18
II-C-25.	Canopus Azimuth Angle . . . . .	II-C-18
II-C-26.	Vega Co-Elevation Angle . . . . .	II-C-19
II-C-27.	Vega Azimuth Angle . . . . .	II-C-19
II-C-28.	Achernar Co-Elevation Angle . . . . .	II-C-20
II-C-29.	Achernar Azimuth Angle . . . . .	II-C-20
II-C-30.	Sirius Co-Elevation Angle . . . . .	II-C-21
II-C-31.	Sirius Azimuth Angle . . . . .	II-C-21
II-C-32.	Nominal Approach Path . . . . .	II-C-22
II-C-33.	Major Mission Events Encke '80 Rendezvous Space Vehicle . . . . .	II-C-24
II-C-34.	Thruster Sequencing for an Equal-Burn Policy . . . . .	II-C-25
II-D-1.	Space Vehicle Configuration . . . . .	II-D-3

## CONTENTS (Contd)

## FIGURES

II-D-2.	Space Vehicle (Exploded View) . . . . .	II-D-5
II-D-3.	Space Vehicle Stowed in Shuttle Bay . . . . .	II-D-8
II-D-4.	Spacecraft Compartment Packaging Arrangement . . . . .	II-D-9
II-D-5.	Space Vehicle Functional Block Diagram . . . . .	II-D-15
II-D-6.	Data Handling System Functional Block Diagram . . . . .	II-D-19
II-D-7.	Command System Functional Block Diagram . . . . .	II-D-19
II-D-8.	Typical Integral Shearplate Chassis Type Electronic Assembly . . . . .	II-D-24
II-D-9.	Typical Plug-in Subchassis Type Electronic Assembly . . . . .	II-D-25
II-E-1.	Viking Based Power Subsystem for SEP Spacecraft . . . . .	II-E-2
II-E-2.	Reaction Control Subsystem Block Diagram . . . . .	II-E-5
II-E-3.	Pitch, Roll and Yaw Control Conceptual Block Diagram . . . . .	II-E-8
II-E-4.	Conceptual Block Diagram for Control of Inertial and Commanded Turns . . . . .	II-E-8
II-E-5.	Telecommunications System Block Diagram . . . . .	II-E-12
II-E-6.	Command Performance Margin . . . . .	II-E-16
II-E-7.	Telemetry Performance Margin . . . . .	II-E-16
II-E-8.	High Rate Telemetry Performance . . . . .	II-E-17
II-E-9.	CCS Hardware Functional Block Diagram . . . . .	II-E-18
II-E-10.	CCS Processor Functional Block Diagram . . . . .	II-E-18
II-E-11.	DSS Block Diagram . . . . .	II-E-26
II-E-12.	Spacecraft Structure . . . . .	II-E-31
II-F-1.	Thrust Subsystem Functional Block Diagram . . . . .	II-F-2
II-F-2.	SEP Module Power Subsystem with Mariner/Viking Spacecraft . . . . .	II-F-8
II-F-3.	Solar Array Configuration . . . . .	II-F-9
II-F-4.	Master Flight Data Subsystem Functional Block Diagram . . . . .	II-F-12
II-F-5.	SEP Module Structure . . . . .	II-F-15
II-F-6.	Pitch Axis Conceptual Block Diagram . . . . .	II-F-19
II-F-7.	Roll or Yaw Axis Conceptual Block Diagram . . . . .	II-F-20

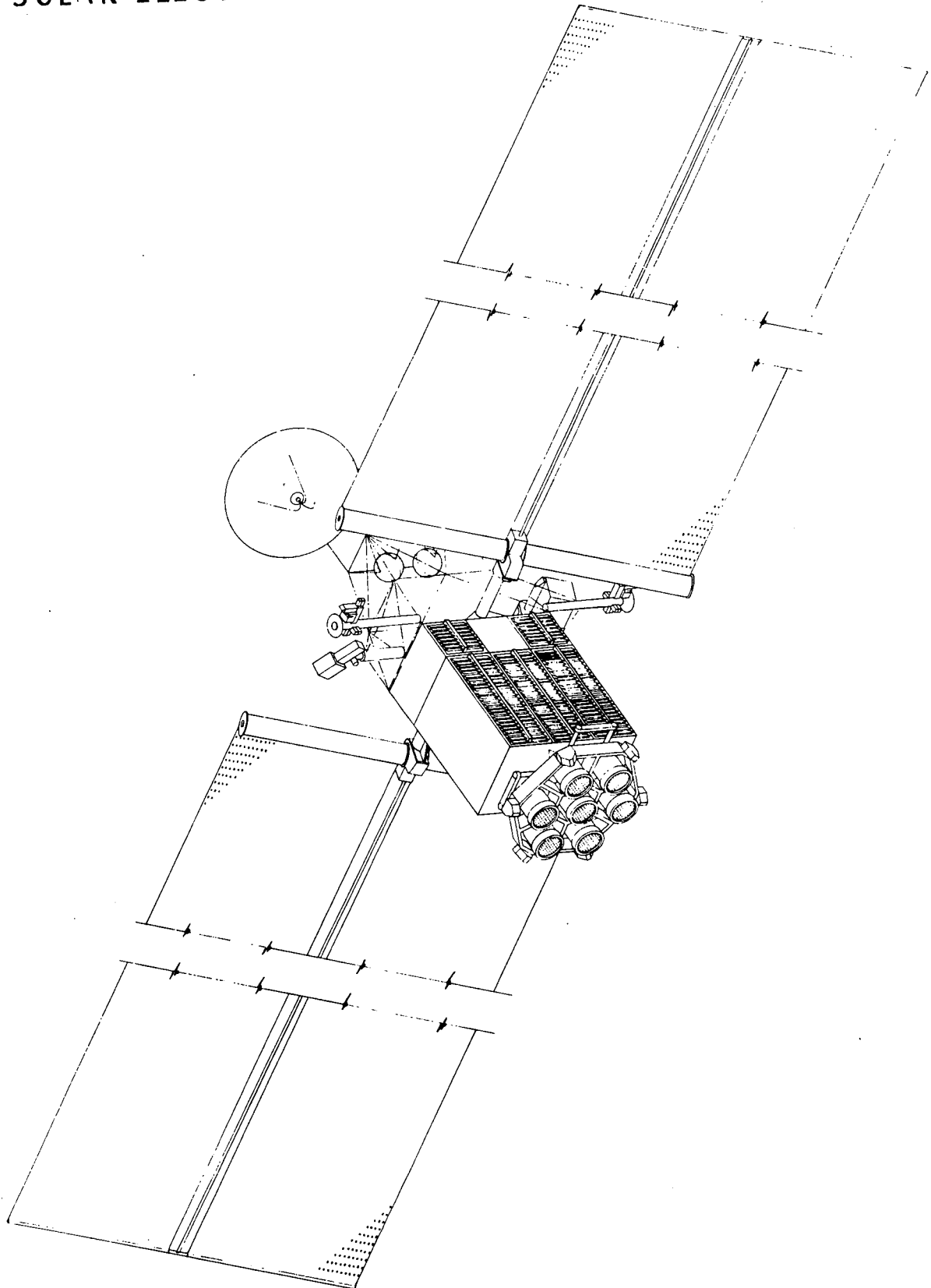
# LIST OF ABBREVIATIONS

A/C	Attitude Control
ACS	Attitude Control Subsystem
A/G	Approach Guidance
bps	bits per second
CABL	Cabling Subsystem
CCS	Computer Command Subsystem
CDU	Command Detector Unit
DEV	Mechanical Devices
DRVID	Differenced Range versus Integrated Doppler
DSIF	Deep Space Instrumentation Facility
DSS	Data Storage Subsystem
DSSE	Data Storage Subsystem Electronics
DSST	Data Storage Subsystem Transport
EOT	End-of-Tape
FDS	Flight Data Subsystem
HGA	High Gain Antenna
ips	inches per second
I/O	Input/Output
IR	Infrared
JPL	Jet Propulsion Laboratory
L	Launch Date
LGA	Low Gain Antenna
MCABL	SEP Module Cabling Subsystem
MDEV	SEP Module Mechanical Devices Subsystem
MDS	Modulation/Demodulation Subsystem
MFDS	Master Flight Data Subsystem
MPSS	SEP Module Power Subsystem
MSTRU	SEP Module Structure Subsystem
NIS	Non-Imaging Science
OD	Orbit Determination
PC	Power Conditioner
PDM	Power Distribution Module

LIST OF ABBREVIATIONS (Contd)

PSS	Power Subsystem
QVLBI	Quasi Very Long Baseline Interferometry
R	Rendezvous
RCVR	Receiver
RCS	Reaction Control Subsystem
RFS	Radio Frequency Subsystem
RING	Remote Interconnecting Group
S	Search
S/C	Spacecraft
SE	Support Equipment
SEP	Solar Electric Propulsion
STRU	Structure Subsystem
SXA	S/X Band Antenna Subsystem
S/V	Space Vehicle
TBD	To Be Determined
TC	Thermal Control
TSS	Thrust Subsystem
TTA	Thruster Translator Assembly
TV	Television
TVC	Thrust Vector Control
TWTA	Traveling Wave Tube Amplifier
VCO	Voltage Controlled Oscillator
VIS	Visual Imaging Science
VO75	Viking Orbiter 1975
XTX	X-Band Transmitter Subsystem

# SOLAR ELECTRIC PROPULSION SPACE VEHICLE





## SECTION I

### INTRODUCTION

This volume describes in detail the solar electric propulsion (SEP) space vehicle and the mission to which it is applied. It includes a detailed functional description of the SEP thrust subsystem along with its technical specifications and requirements as are known at this time. Detailed analyses which were performed in support of the SEP module thrust subsystem functional description document are reported in Volume III of this report series.

Volume I of this report series contains a technical summary of the work documented in Volumes II and III.

The Encke Rendezvous mission was chosen to provide a framework for the space vehicle design because it is one of the most difficult SEP-type missions to perform. It would thus furnish a pacing goal to stress SEP technology development. A design that would be successful in this mission could be used, with minor modifications, in a class of missions less demanding, although scientifically rewarding.

Because the Thrust Subsystem Detailed Functional Description (Appendix A) is the first attempt at writing such a document, and because it is expected to be revised and updated as development progresses, it has been presented in the form of a prototype functional requirement. Two major applicable documents, packaging and cabling design requirements, are contained in Appendices B and C, respectively. These are unreleased copies, marked "Information Only" and are subject to revision.

An explanation of some study-peculiar terminology will aid the reader to understand the subsequent descriptions. The electric propulsion thrust subsystem and its supporting subsystems, including the roll-out solar arrays, comprise the SEP MODULE. This module is attached to a SPACECRAFT which contains science instruments, and Viking-type subsystems. The combination of both SPACECRAFT and SEP MODULE is called the SPACE VEHICLE. A pictorial representation of this nomenclature is shown in Fig. I-1.

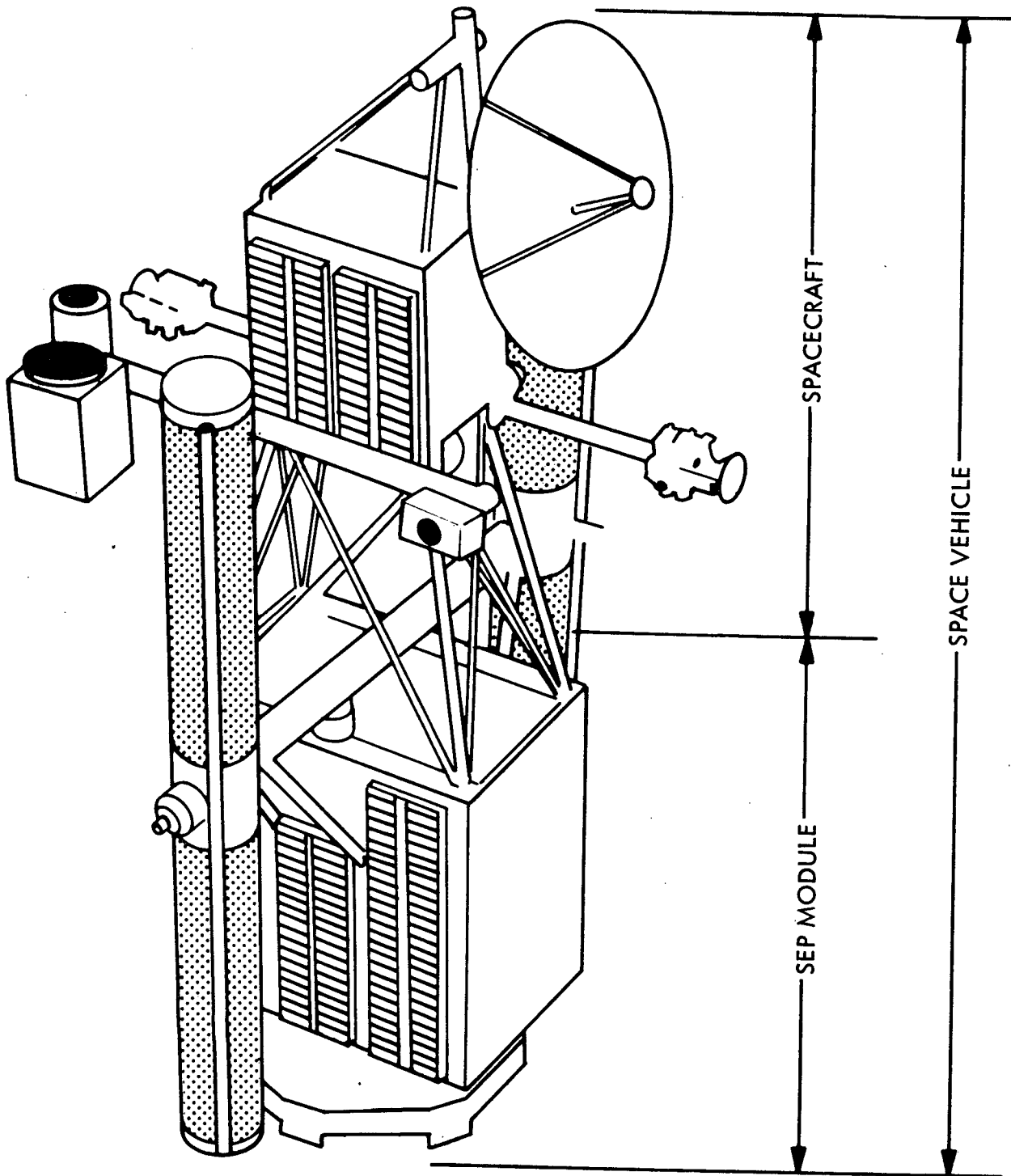


Figure I-1. Space Vehicle.

## SECTION II

### FUNCTIONAL DESCRIPTION

#### A. SCOPE

The purpose of this document is to provide a description of the baseline space vehicle system which was utilized to support SEP thrust subsystem functional description studies. Emphasis will be placed on those areas which impact the thrust subsystem description; however, most areas of the mission and space vehicle systems will be treated for completeness.

B. REQUIREMENTS AND CONSTRAINTS

The following are mission requirements and constraints:

- (1) A mission to rendezvous with the comet P-Encke will be performed with an SEP module attached to a spacecraft consisting primarily of Viking subsystems.
- (2) Modifications to the Viking Orbiter 1975 (VO75) spacecraft subsystems should be minimized.
- (3) The SEP module should provide for all functions required by the SEP mission that are beyond VO75 capability.
- (4) The thrust subsystem portion of the SEP module must be jettisonable, leaving a viable spacecraft after the propulsion phase of the mission.
- (5) Science requirements will not be considered, although a typical comet science package will be included in mass estimates.
- (6) Rendezvous is defined as 1000 km in position and 4.0 m/sec in velocity relative to Encke. Post-rendezvous operations were not considered.
- (7) Rendezvous must be achieved prior to 40 days preceding Encke's perihelion.
- (8) Sufficient command margins on spacecraft omni-antenna with the DSN 210-foot antenna should be maintained throughout the mission.
- (9) Telemetry and data system performance will be sufficient to support engineering and approach navigation requirements throughout the mission.
- (10) The spacecraft will be configured to be compatible with the Shuttle-Centaur launch system and the Titan IIID/Centaur with a 14-foot shroud.

## C. ENCKE '80 RENDEZVOUS MISSION DESCRIPTION

### 1. Trajectory Characteristics

Trajectory options for the 1980 Encke rendezvous mission were investigated to determine a nominal flight path for use in hardware analyses and trade studies. Primary considerations included launch date, flight time, and arrival date. Objectives were:

- (1) Sufficient mass capability for a vehicle of 1200 to 1300 kg (final mass,  $M_F$ ).
- (2) Reasonably fast transfer to the comet.
- (3) Early arrival at the comet to provide flexibility for exploration strategies.
- (4) Achievement of these objectives with a 16 kw power allowance for the thrust subsystem ( $P_O$ ).

The thrust subsystem was assumed to be capable of a fixed 3,000-second specific impulse at an efficiency ( $\eta$ ) of 0.62.

The normalized solar power profile as a function of solar distance is given in Fig. II-C-1. In the trajectory analyses, this profile was adjusted to provide an auxiliary power allowance,  $\Delta P/P_O \approx 0.02$ . No adjustments were included to account for discretizing the solar array orientation angle for normal sun incidence. The characteristic mission times, plus the general geometry of the angle between sun line and thrust beam, along with predicted hardware orientation capability, should allow a close approximation of the power profile.

Figure II-C-2 illustrates flight time trade data for direct (transfer angles less than 360 deg) trajectories. Although indirect trajectories were also considered, the characteristic flight time range for them begins at about 1,100 to 1,200 days. The longer times, plus the feature of initial passage inside the earth's orbit made selection of an indirect nominal trajectory unattractive,

even though increased mass capability is available. As shown in Fig. II-C-2, direct missions in the 950- to 1,000-day range appear to provide sufficient capability with an arrival in the comet vicinity 50 days before its perihelion passage.

Launch period alternatives were examined for several criteria, including the desirability of planned coasts for performance contingency. Because of the unavailability of adequate simulation software, designed coasts could not be included. While optimally placed coasts could have been included in the nominal trajectory, and, in fact, were considered in individual trade studies, the path most emphasized was one without coasts in order to provide stringent requirements for hardware considerations. This approach stressed "worst-case" conditions and, therefore, the nominal trajectory described is qualified as "preliminary" and must be updated as detailed mission design proceeds. However, the eventual designed path will not be vastly different in overall geometry.

Generally, the trajectories of interest correspond to launch dates from mid-February to slightly past mid-March in 1978. The trajectory selected for hardware implementation analysis, risk evaluation, and navigation studies begins on 16 March 1978. Transfer time is 950 days. The arrival date is 21 October 1980, 47 days before the comet's perihelion passage (6 December 1980). The trajectory's ecliptic projection is given in Fig. II-C-3, showing positions of earth and vehicle every 100 days. At earth, the corresponding injection energy parameter,  $C_3$ , is  $54 \text{ km/sec}^2$ , which is typical of the range from 20 to  $100 \text{ km/sec}^2$ , where SEP performance is relatively insensitive to injection energy.

For the selected path, the available solar power history is given in Fig. II-C-4. This profile constitutes the basis for thrust subsystem design and power matching policy.

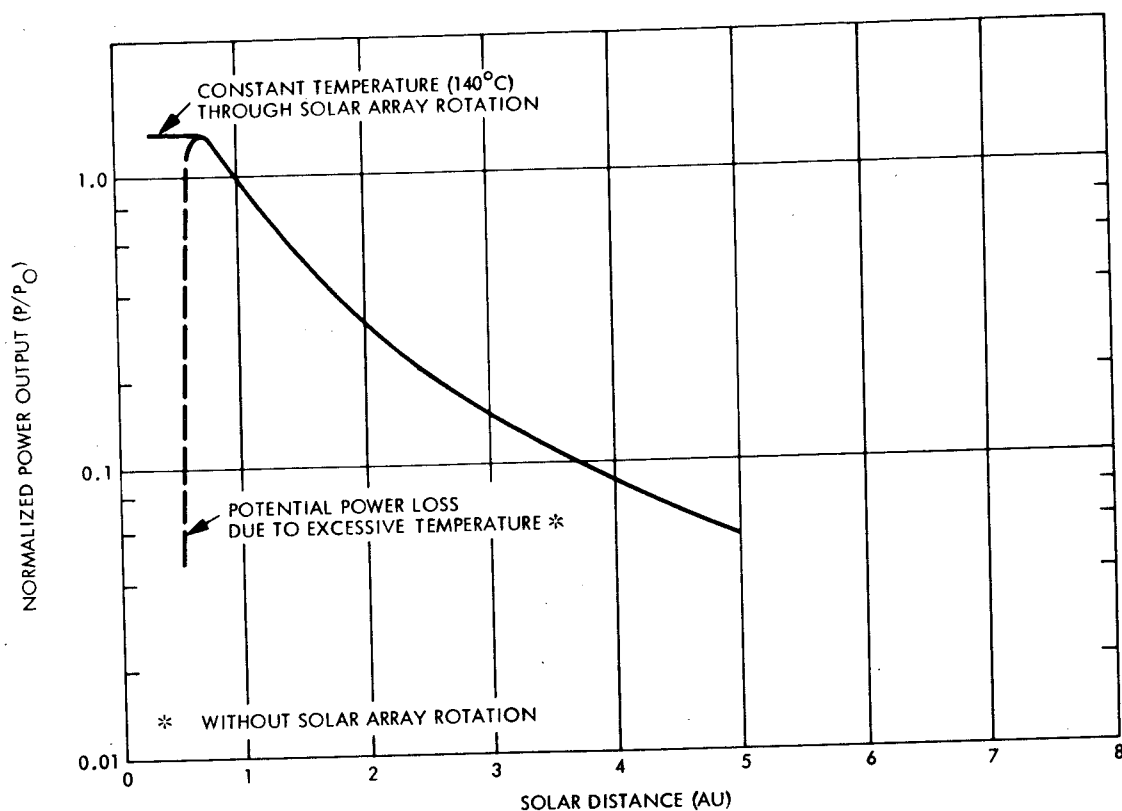


Fig. II-C-1. Normalized Solar Power Profile

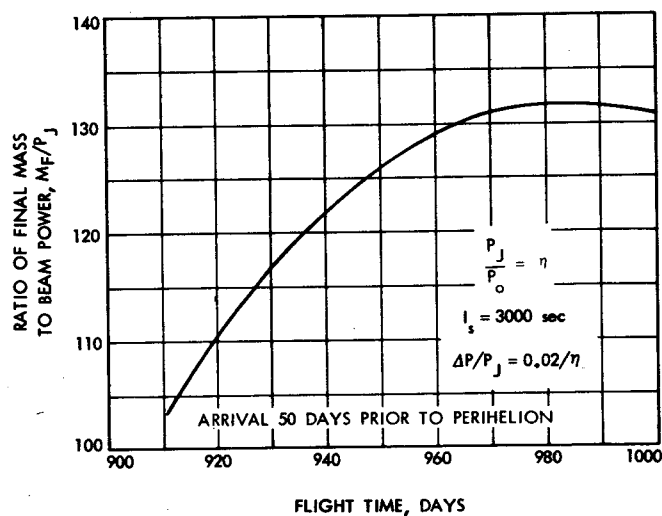


Fig. II-C-2. Flight Time and Performance Trade Data for Direct Rendezvous Trajectories to Encke (1980 Perihelion)

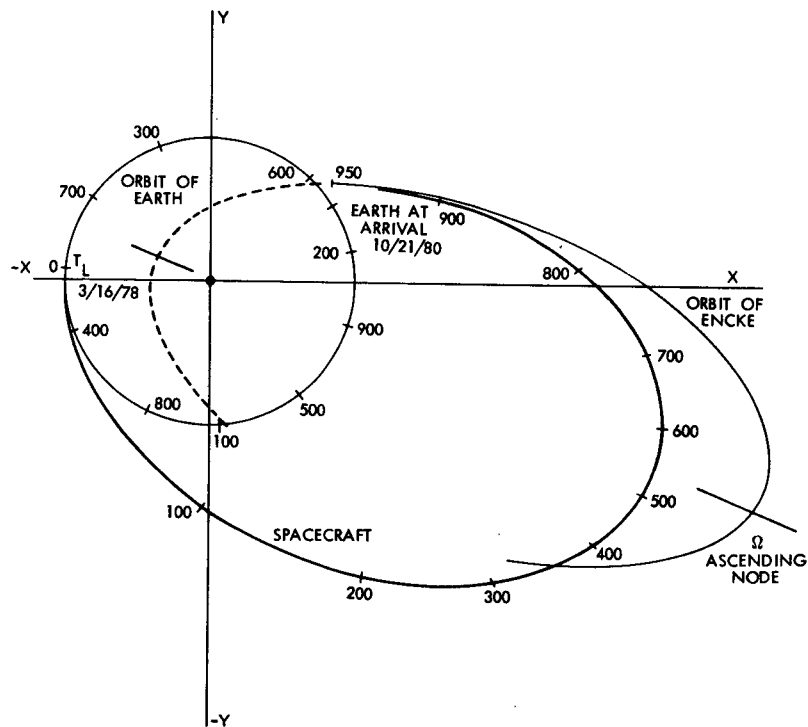


Fig. II-C-3. Encke Rendezvous (1980) Nominal Trajectory  
Ecliptic Projection

For an adequate description of the nominal trajectory, an explanation of the time-varying thrust profiles common to solar electric missions is needed. The principal thrust component applied in the orbital plane must initially be along the path to increase the aphelion radius. Subsequently, as the vehicle reaches aphelion, the in-plane component must be directed retrograde to decrease the perihelion radius. To reach Encke's small perihelion radius (0.34 AU), a large total retroimpulse must be applied near aphelion where the solar electric power available is only 10 to 20 percent of its value near earth. This explains the large, initial power requirement attributed to the Encke rendezvous mission. After aphelion, the component is again directed posigrade to further increase aphelion until the comet's orbit is matched.

Out-of-plane thrusting is also required to match Encke's 12.0 deg orbital inclination. The most effective use of this component is near the line



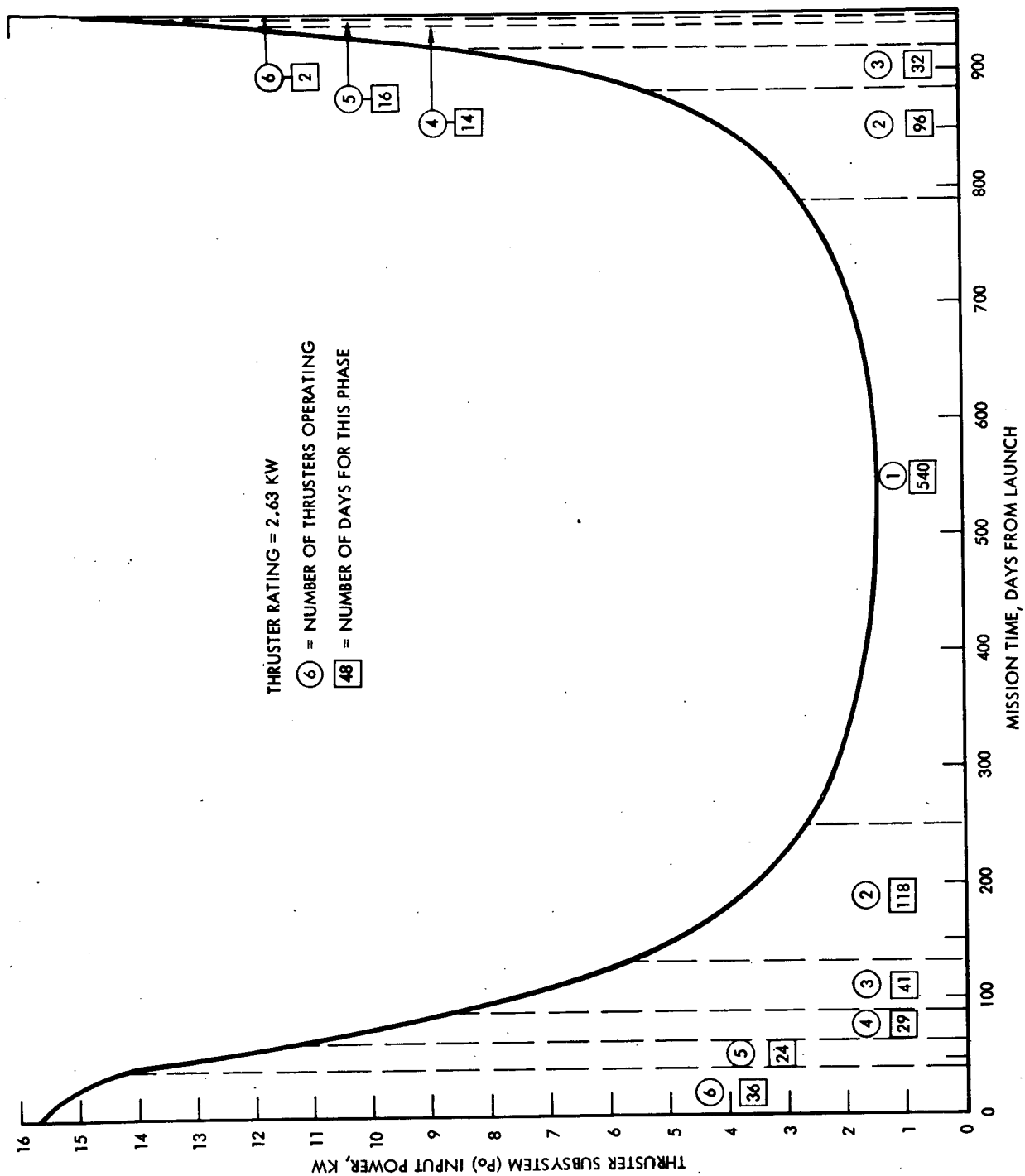


Fig. II-C-4. 1980 Encke Rendezvous Mission, Thruster Subsystem Power Profile for Nominal Trajectory

of nodes between the departure plane (ecliptic) and Encke's orbit. The ascending node of Encke's orbit lies near its aphelion, as shown in Fig. II-C-3. The thrust subsystem is starved for power as the transfer trajectory nears the line of nodes outbound, reinforcing the requirement for large, initial power supplies on this mission.

To facilitate analysis of time-varying pointing requirements for the thrust vector, a body-fixed, vehicle-centered coordinate system was adopted. The need for such a system is due to the continuous change in the thrust pointing with respect to the sun. This changing thrust program causes reference stars, earth, and sun to change location in the vehicle coordinate system and makes reference look-angles difficult to define in terms of traditional coordinate systems in which the sun is fixed in vehicle coordinates. The adopted body-fixed system, shown in Fig. II-C-5, orients the +X-axis in the direction of the thruster exhaust beam and the Z-axis normal, with +Z being in the general direction of the sun. Positive Y completes the right-hand system and extends along the solar array rotation axis. Pointing angles in this system are denoted as co-elevation and azimuth. Azimuth is measured in the X-Y plane from the exhaust beam axis, +X. The positive sense is clockwise when looking along the +Z-axis. Note that +Z does not necessarily coincide with the line between the vehicle and the sun. The thrust exhaust vector is always fixed at 0 degrees azimuth and 90 degrees co-elevation in this system.

To define the thrust-pointing history for the nominal trajectory, two angles are used. The first, the sun-vehicle-thrust exhaust beam angle, defines the thrust plane and the thrust direction in that plane. The second angle gives the orientation of the thrust plane about the sun line. The time histories of these angles are given in Figs. II-C-6 and II-C-7. In Fig. II-C-7, the inclination, or roll angle, is measured counter clockwise from the easterly direction on the celestial sphere. A clarifying sketch is given in Fig. II-C-8 which shows the thrust-beam orientation at SEP thrust initiation. Note that Fig. II-C-6 gives the thrust beam angle as 100 degrees and Fig. II-C-7 places the thrust vector (opposite the beam direction) at 330 degrees of roll.

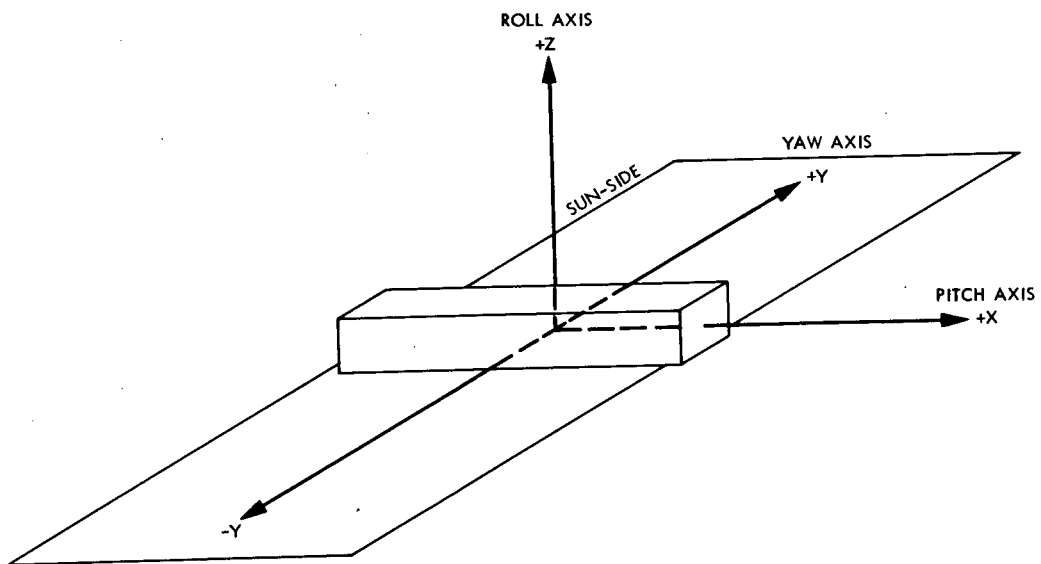
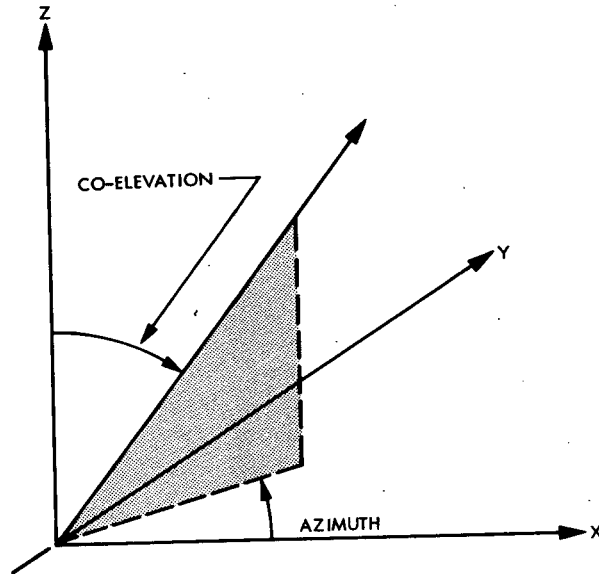


Fig. II-C-5. Body-Fixed Coordinate System for Solar Electric Vehicle

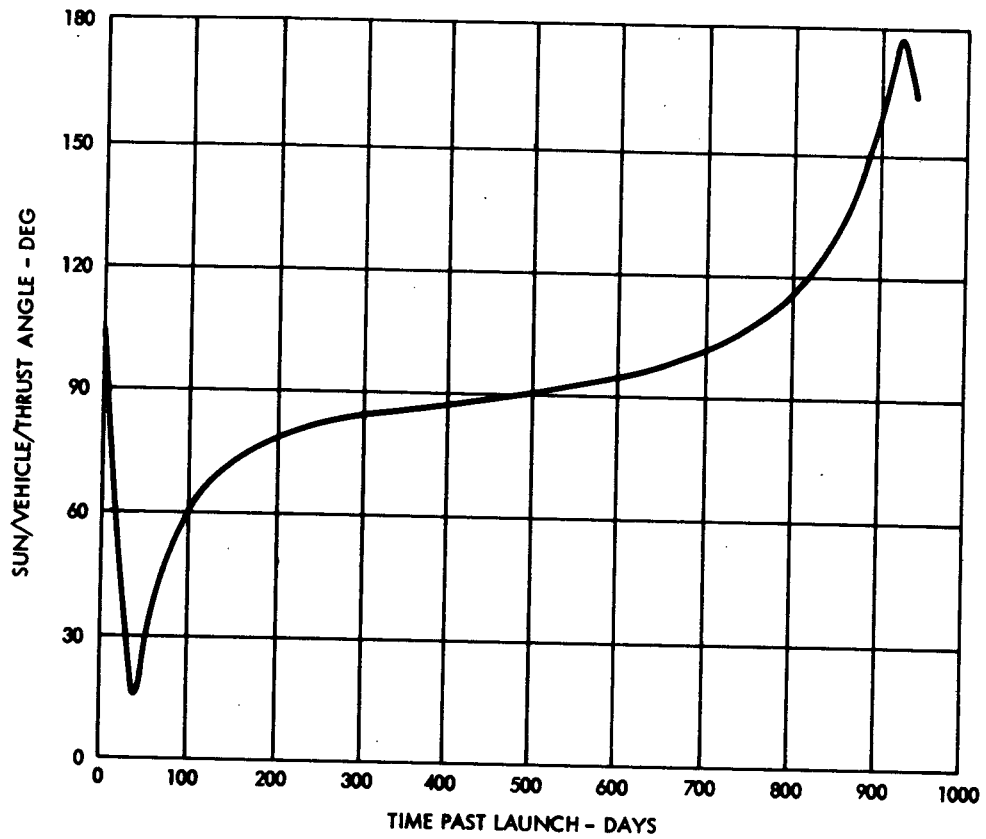


Fig. II-C-6. Sun-Vehicle-Thrust Beam Angle

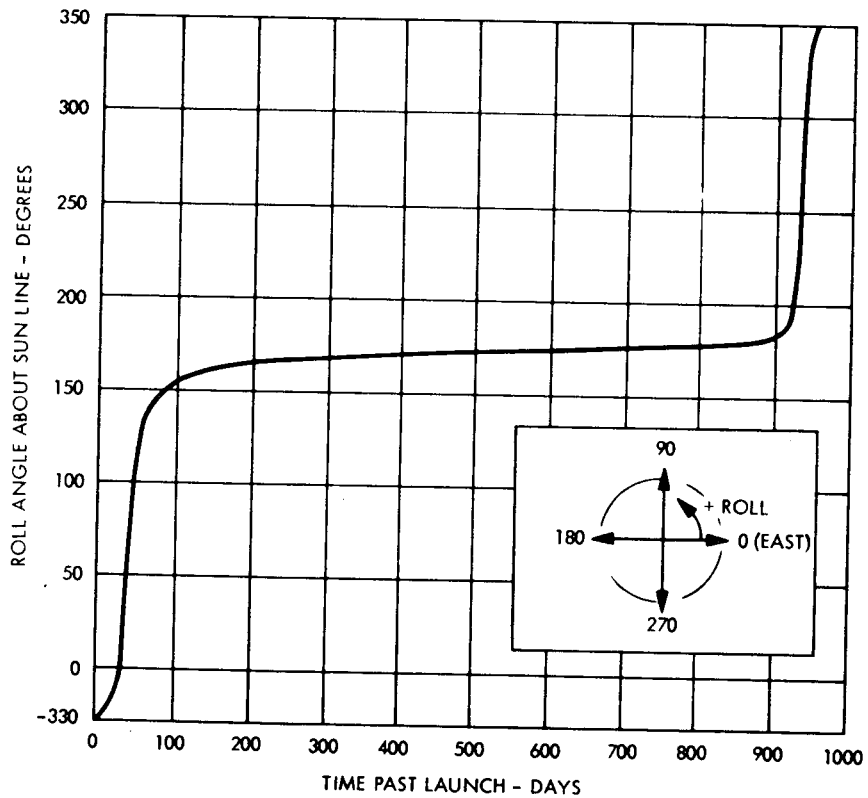


Fig. II-C-7. Thrust Plane Roll Angle

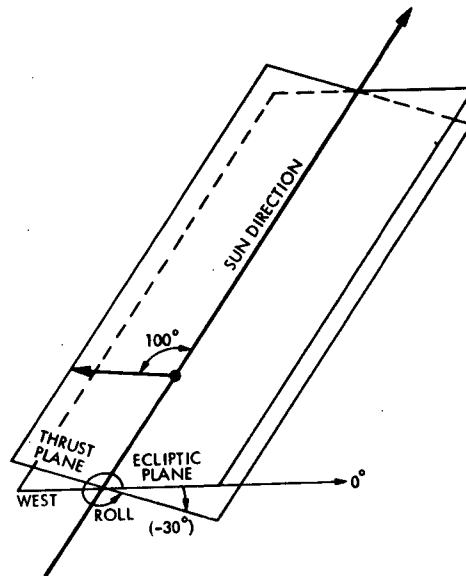


Fig. II-C-8. Thrust Beam Orientation at Thrust Initiation

Figures II-C-9 through II-C-31 comprise a compendium of time histories of other parameters of interest over the 950-day trajectory. These cover ranges, range rates, declinations, propellant mass, and various angles.

The final period before encounter, approximately 100 days, is the most important phase. Up until this point, the vehicle is primarily in a cruise configuration, navigating within the earth-based uncertainty on the comet's expected position. During the cruise, navigational updates once per week should be sufficient. However, as the encounter nears, the knowledge of the comet's position will quickly improve after acquisition by on-board optical sensors. A nominal linear terminal maneuvering strategy was constructed, based on an initial uncertainty of about 30,000 km in Encke's position. Optical on-board recovery of the comet was determined to occur some 60 days preceding the rendezvous. Navigation must begin earlier than 40 days prior to rendezvous. Figure II-C-32 shows the nominal approach path in comet-centered

coordinates. A successful rendezvous concluding this path is defined as a state within 1,000 km of the nucleus with a relative speed of less than 4.0 m/sec. Note that this nominal was selected to provide a stringent test of proposed terminal maneuver strategies and navigation techniques. A relaxation of the rendezvous definition will ease implementation constraints, but this must be done in relation to scientific objectives.

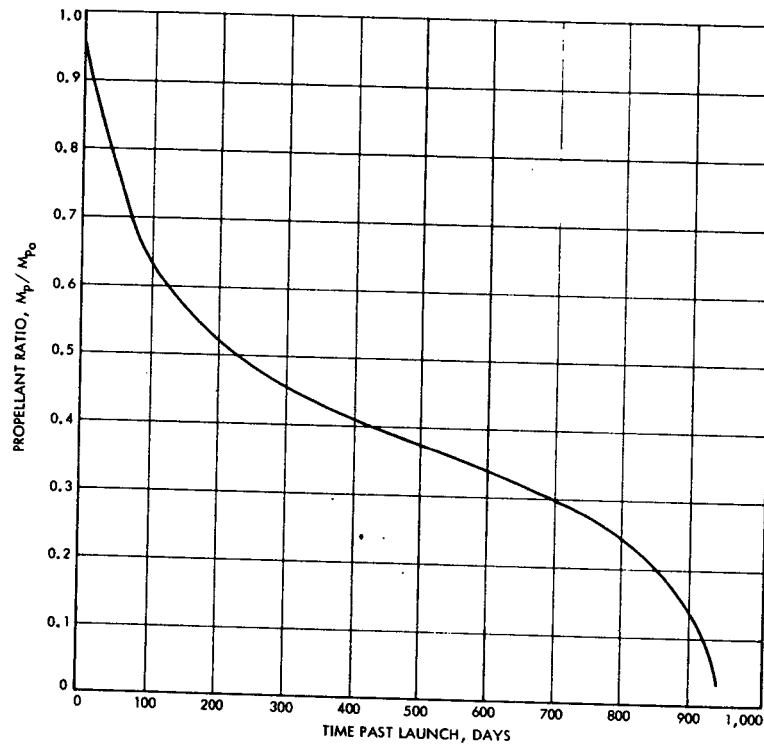


Fig. II-C-9. Propellant Consumption for Nominal Trajectory

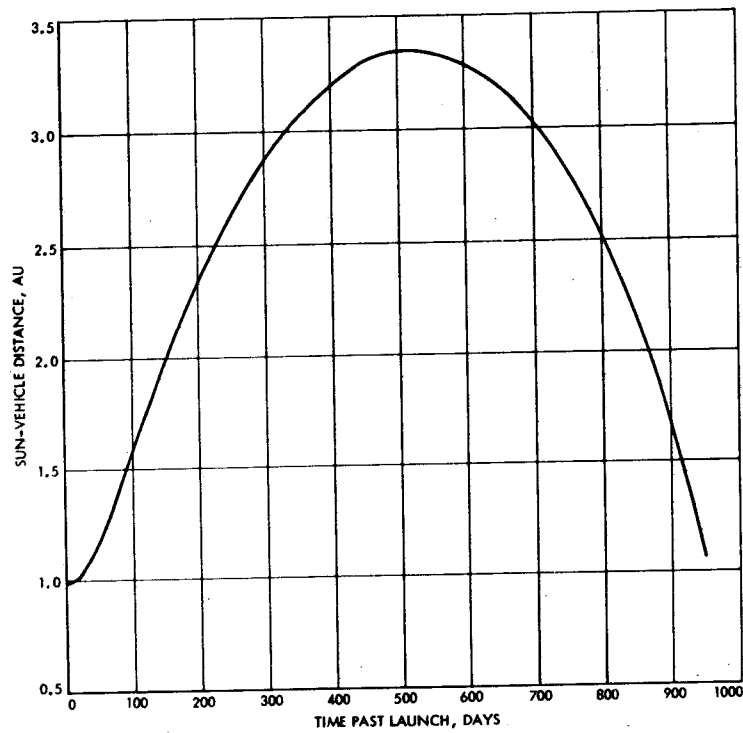


Fig. II-C-10. Sun-Vehicle Distance

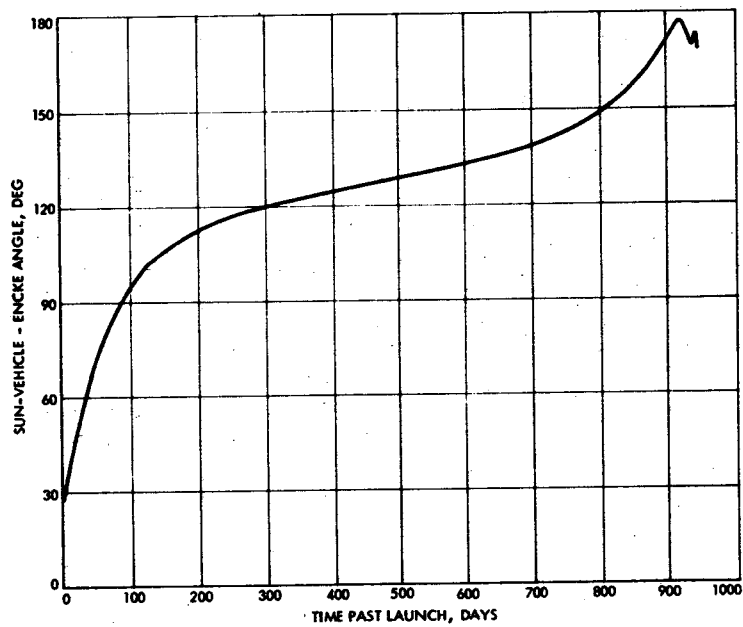


Fig. II-C-11. Sun-Vehicle-Encke Angle

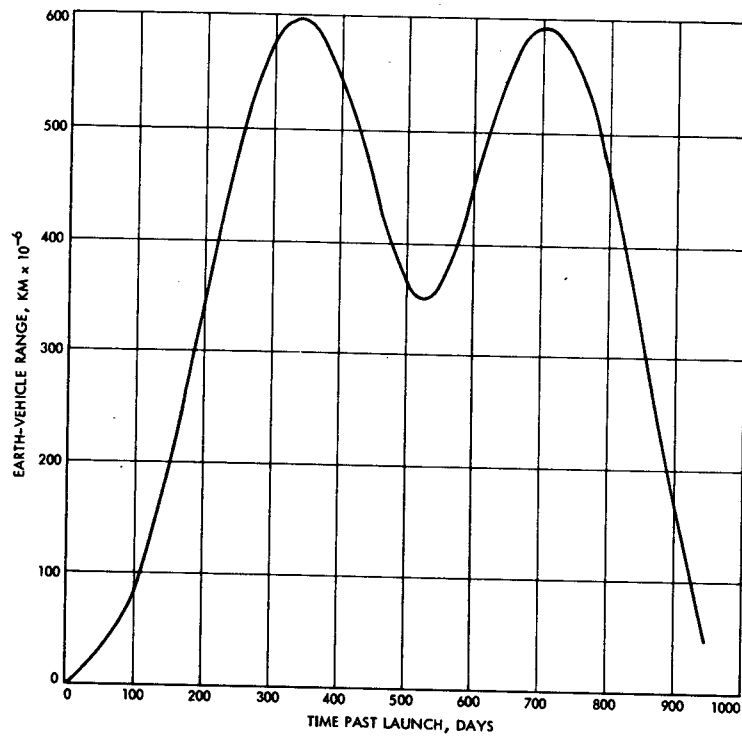


Fig. II-C-12. Earth-Vehicle Range

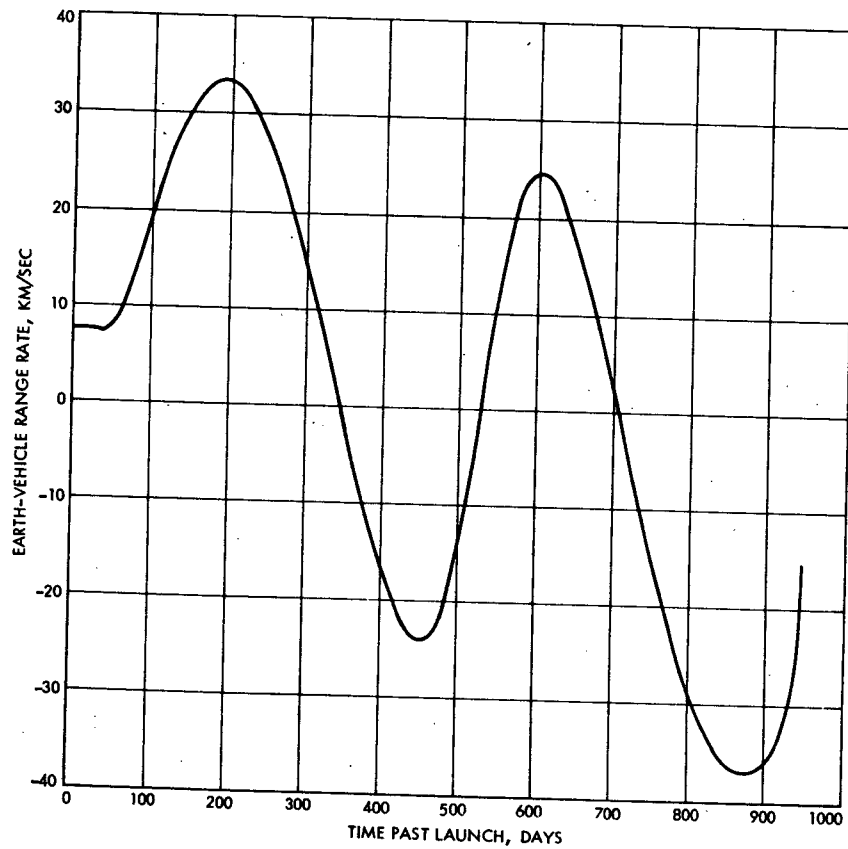


Fig. II-C-13. Earth-Vehicle Range Rate



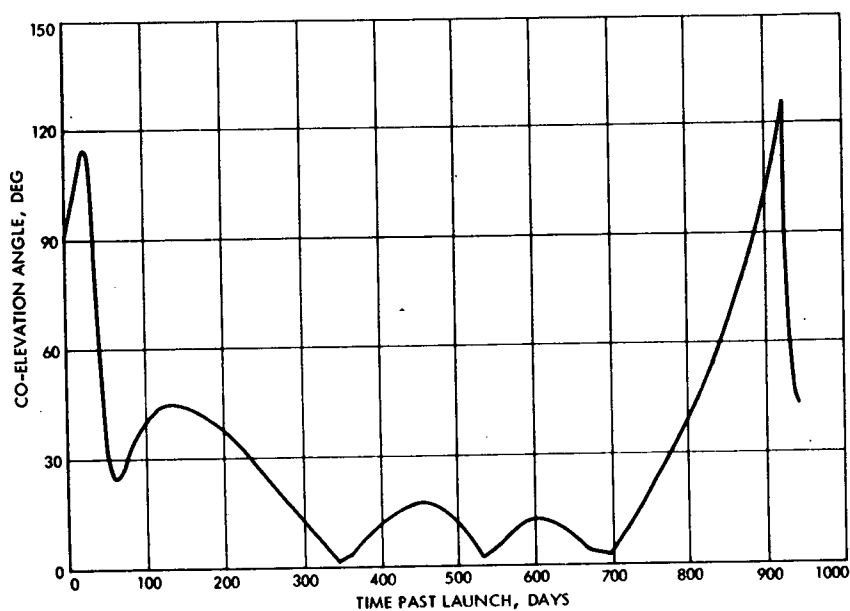


Fig. II-C-14. Earth Co-Elevation Angle

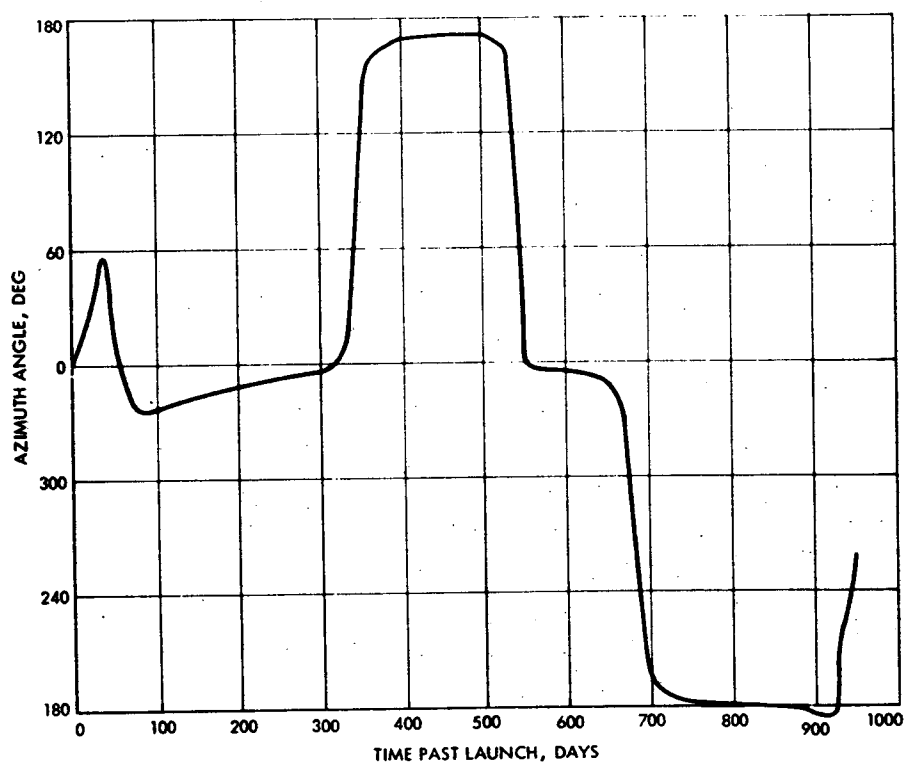


Fig. II-C-15. Earth Azimuth Angle

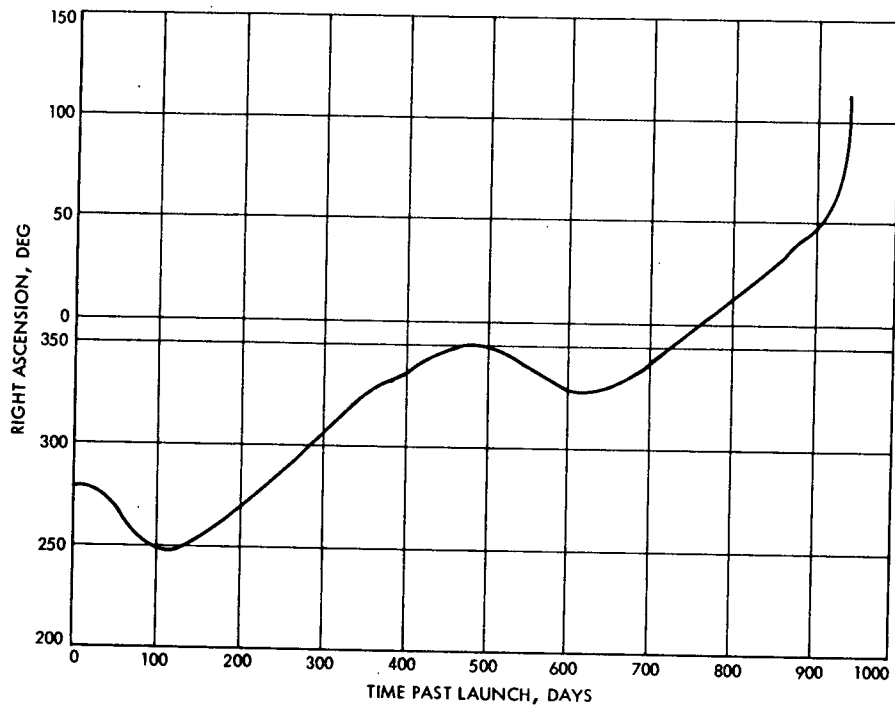


Fig. II-C-16. Vehicle Right Ascension from Earth

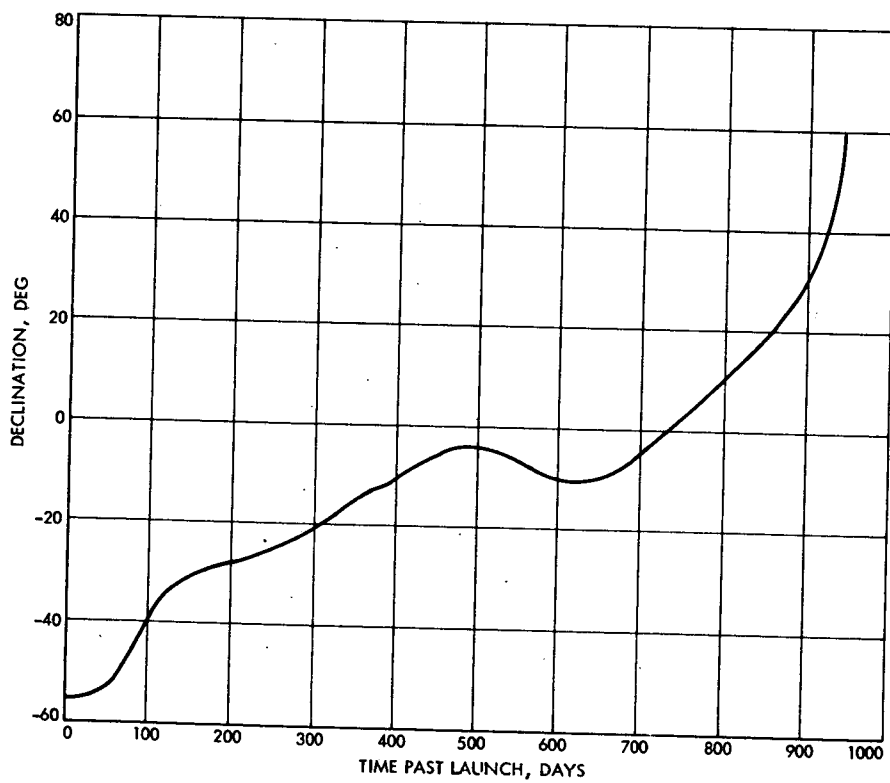


Fig. II-C-17. Vehicle Declination from Earth

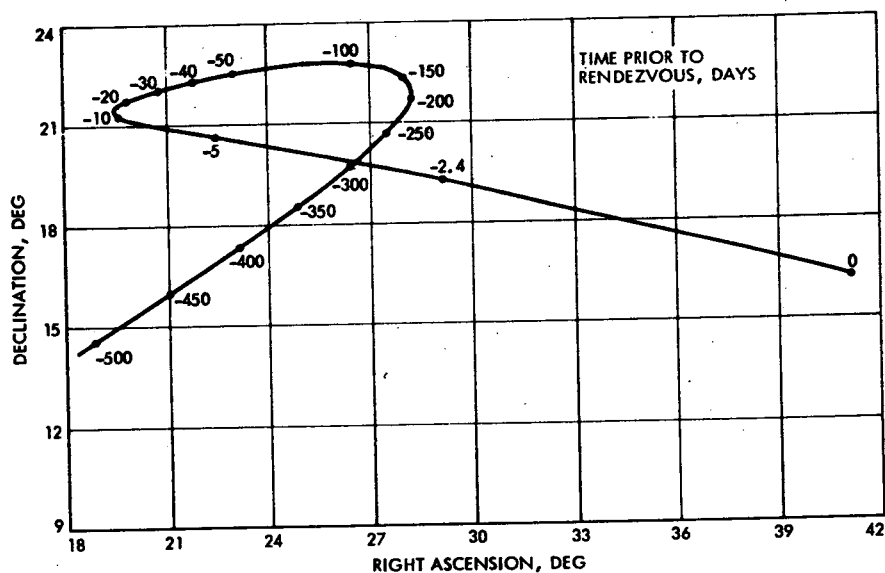


Fig. II-C-18. Encke Right Ascension and Declination from Vehicle

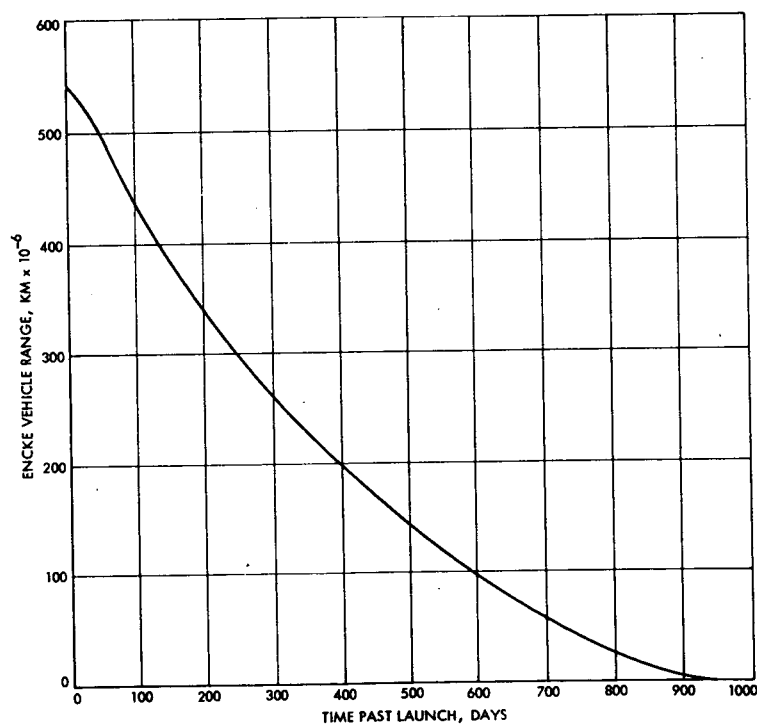


Fig. II-C-19. Encke-Vehicle Range

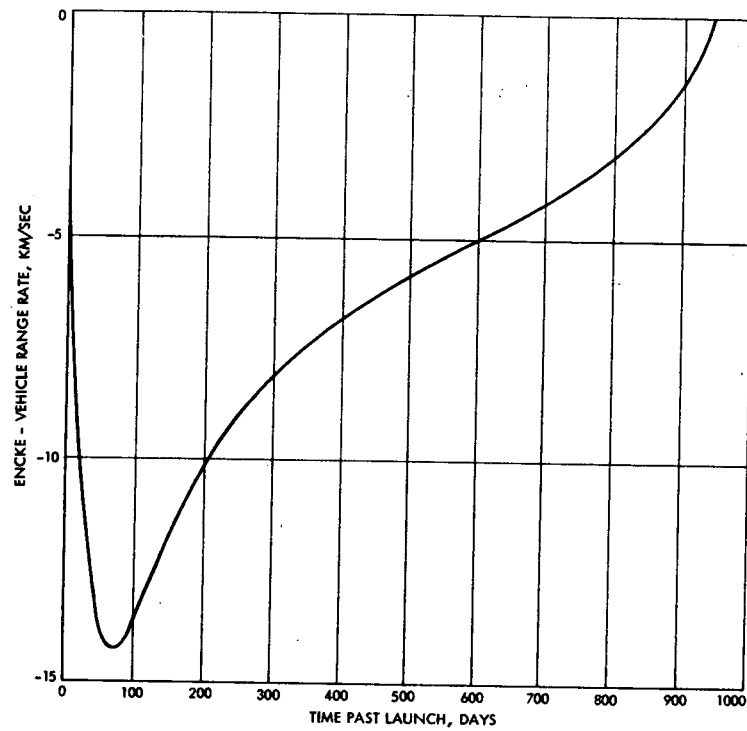


Fig. II-C-20. Encke-Vehicle Range Rate

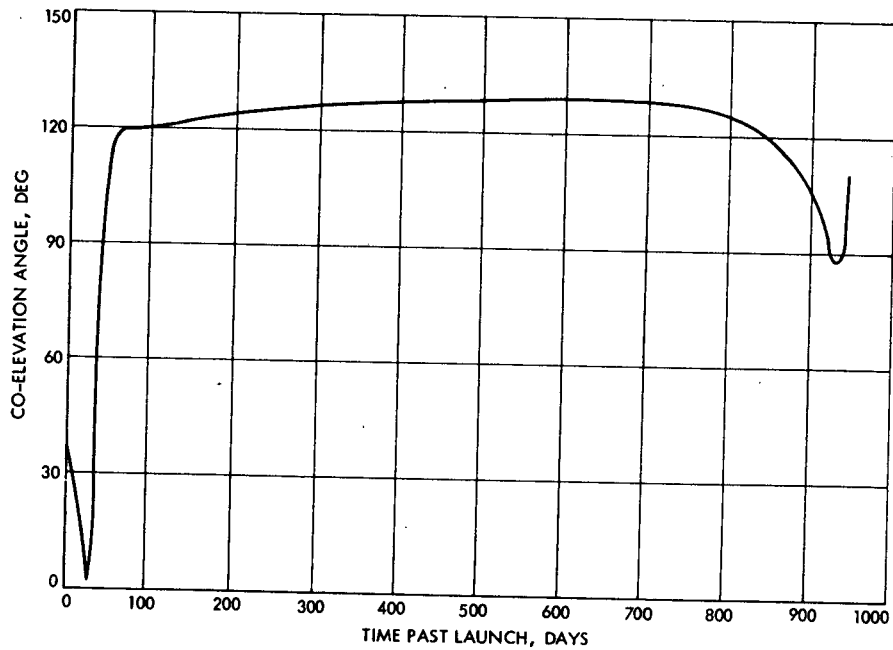


Fig. II-C-21. Encke Co-Elevation Angle

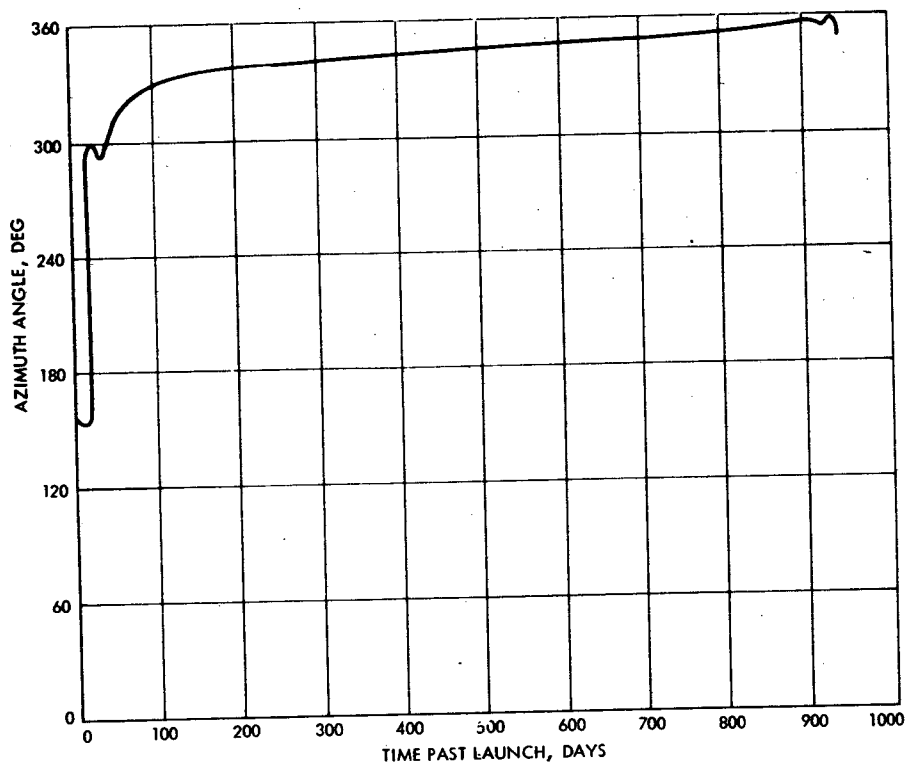


Fig. II-C-22. Encke Azimuth Angle

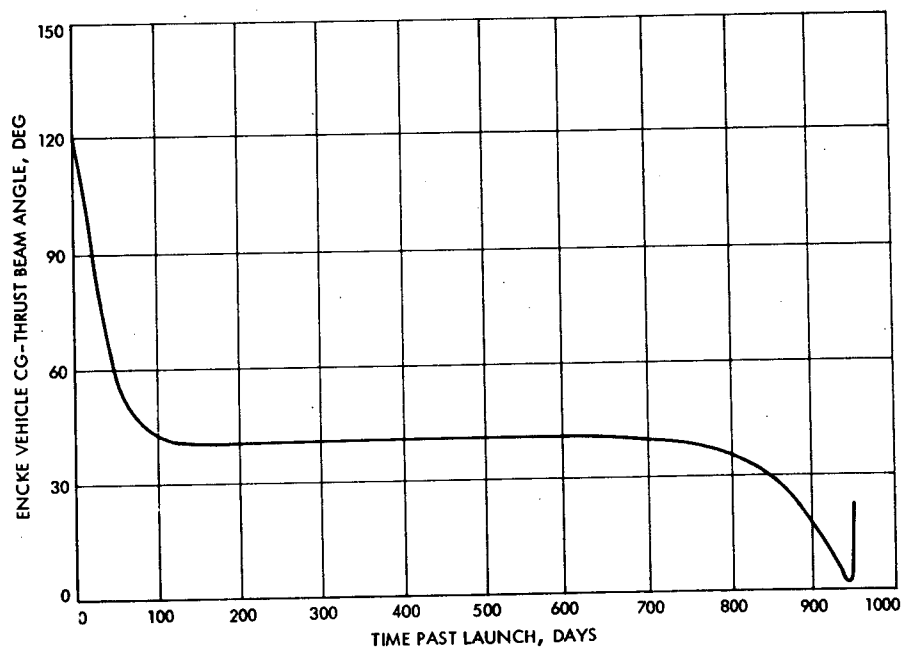


Fig. II-C-23. Encke - Vehicle - Thrust Beam Angle

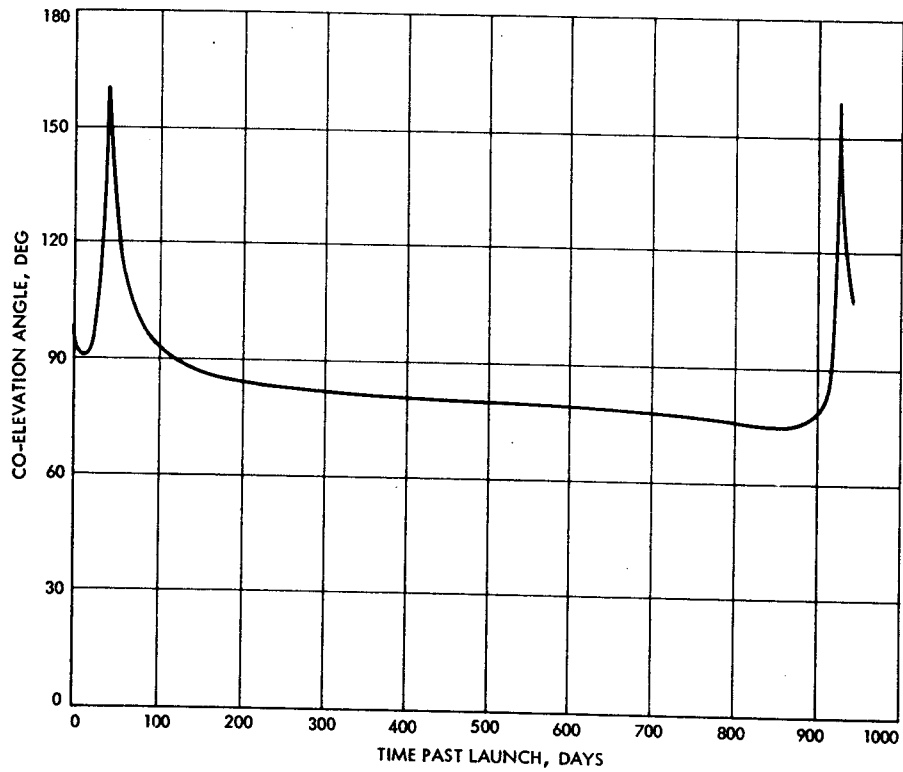


Fig. II-C-24. Canopus Co-Elevation Angle

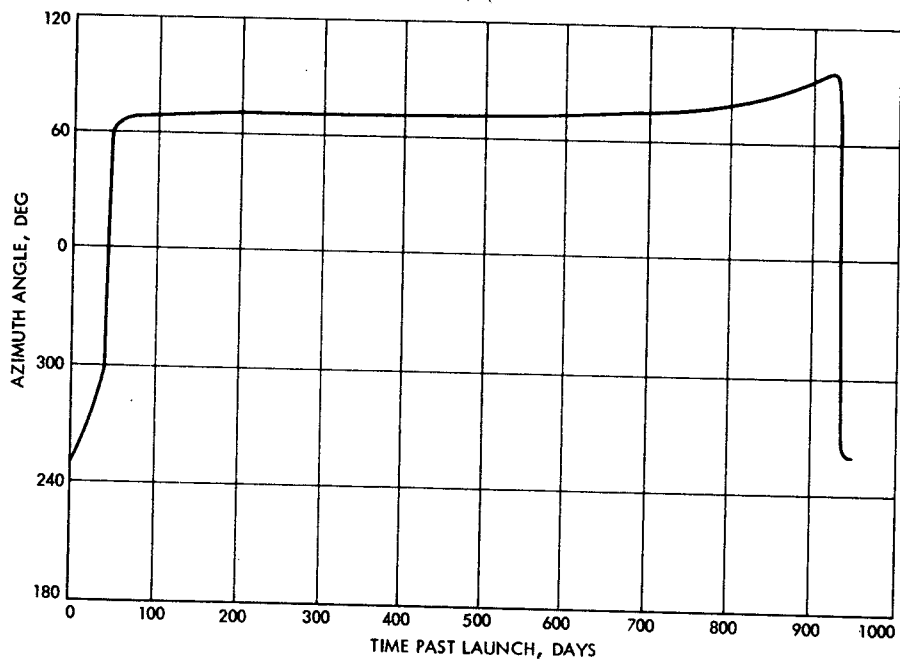


Fig. II-C-25. Canopus Azimuth Angle

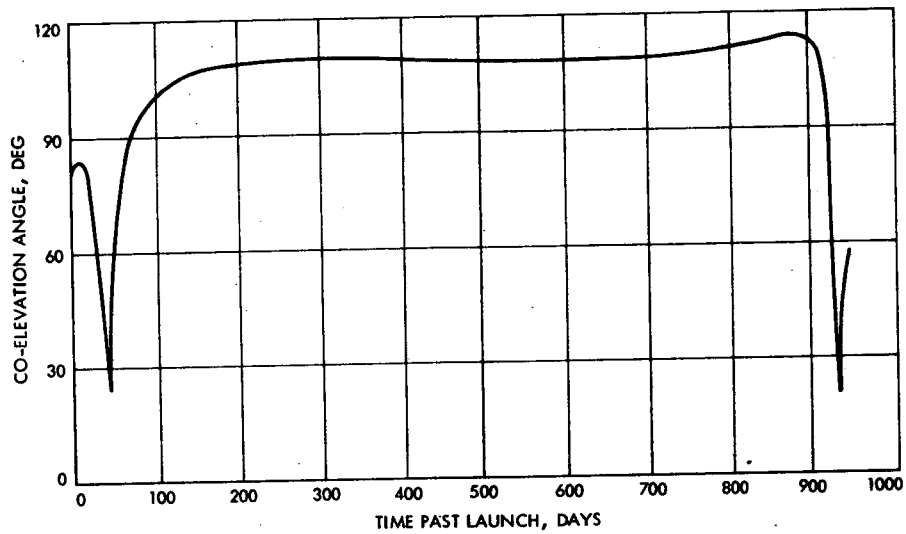


Fig. II-C-26. Vega Co-Elevation Angle

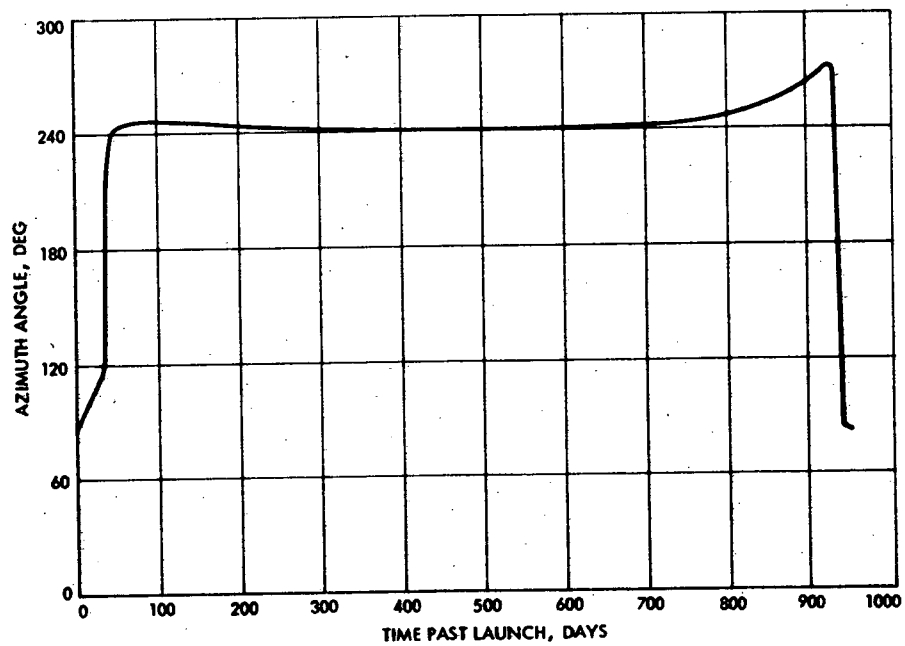


Fig. II-C-27. Vega Azimuth Angle

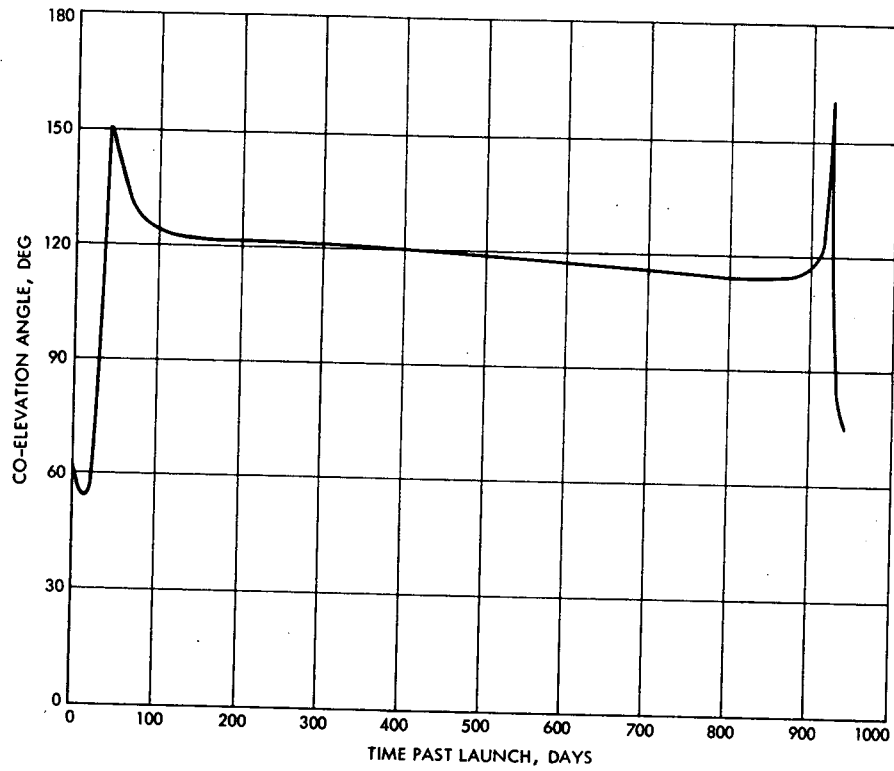


Fig. II-C-28. Achernar Co-Elevation Angle

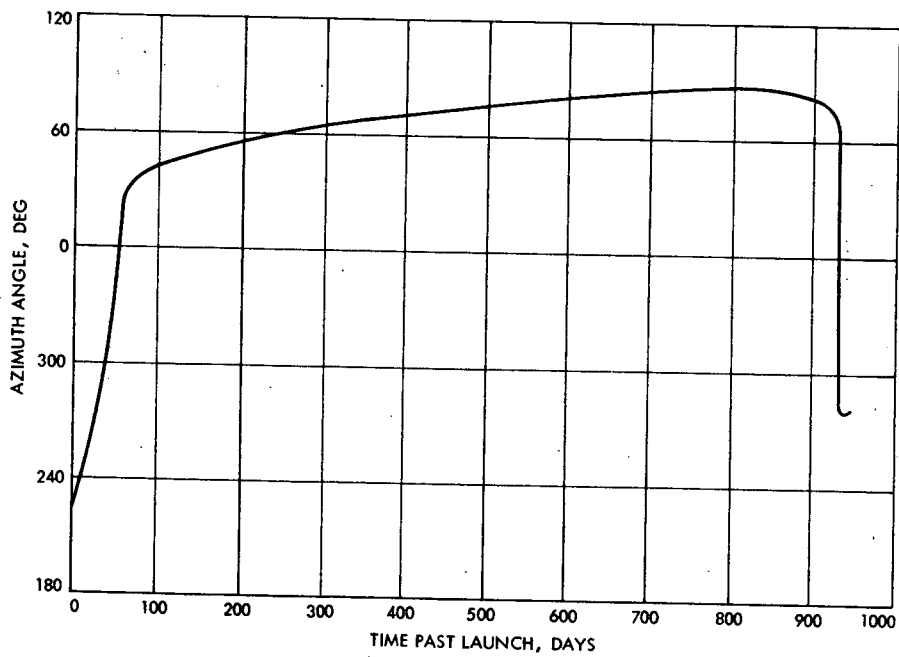


Fig. II-C-29. Achernar Azimuth Angle



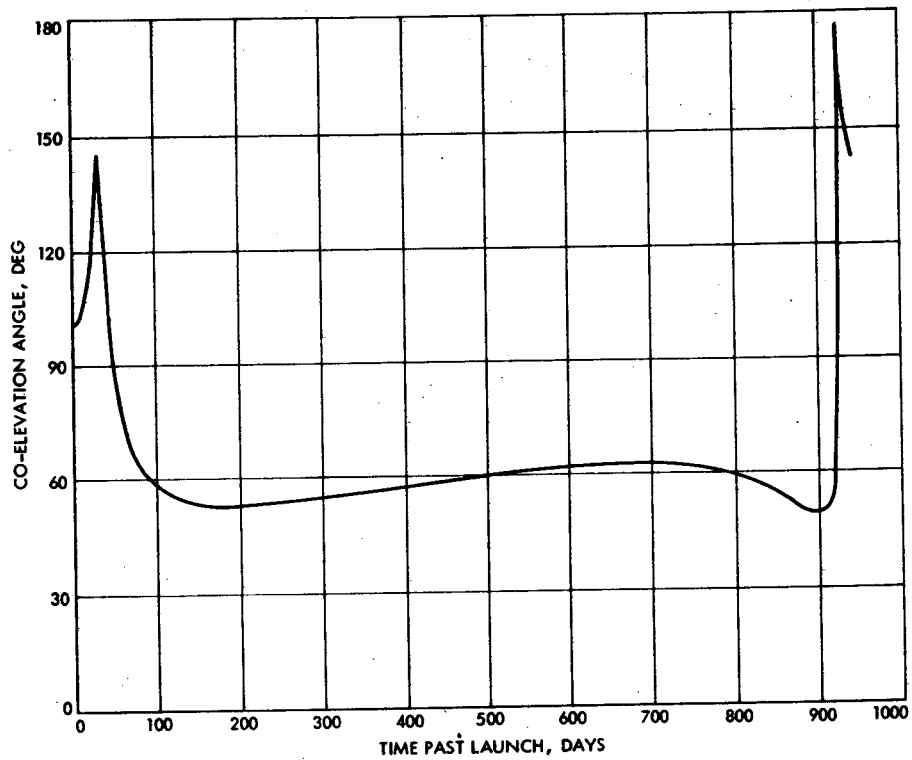


Fig. II-C-30. Sirius Co-Elevation Angle

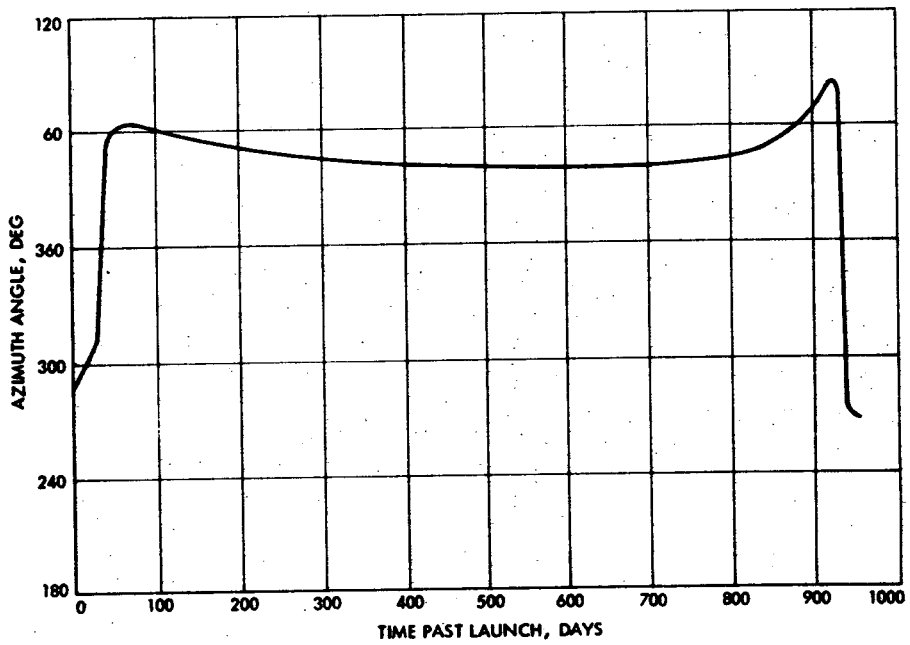


Fig. II-C-31. Sirius Azimuth Angle

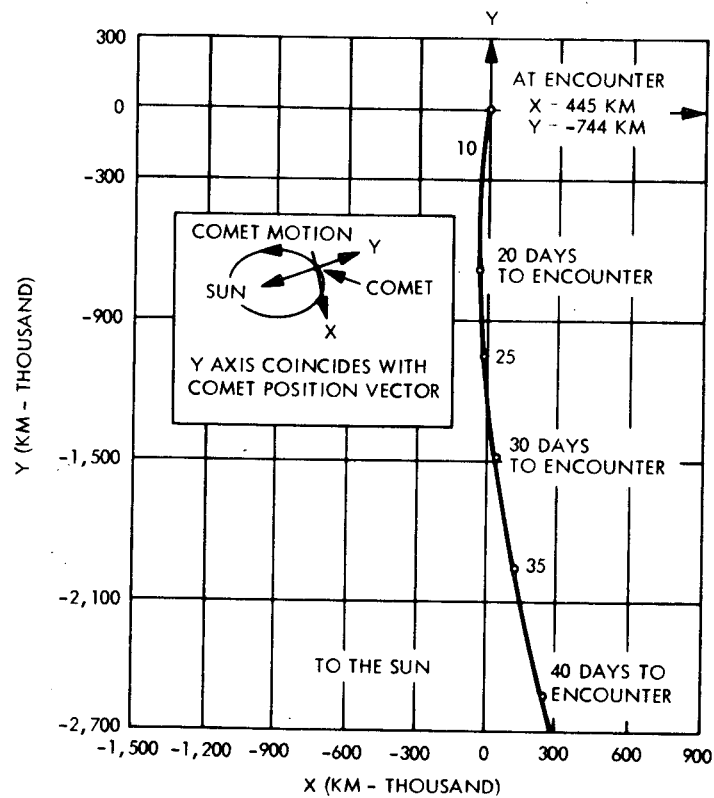


Fig. II-C-32. Nominal Approach Path

2. Sequence of Events

The date of launching the space vehicle is assumed to be March 16, 1978. The sequence of events describes major events occurring from launch until Encke Rendezvous, in chronological order, as shown in Fig. II-C-33. The detailed events follow a description of the five phases:

a. Launch Phase

The launch phase starts with liftoff, and continues through sun and star acquisitions until the cruise phase begins two days later with thruster turn-on. Launch-vehicle-related events have not been included because a specific launch vehicle has not been chosen. Either a Titan III D-Centaur or Space Shuttle-Centaur launch vehicle are possible choices.

b. Cruise Phase

The cruise phase begins with a period of continuous tracking to evaluate thrust subsystem performance in a space environment. Some 11 days later, tracking is performed only as often as necessary to determine the orbit and make the required changes to the thrust vector. This is approximately one pass per week with one 64-m net station. Changes in reference stars, and changing thrust levels made necessary with the changing solar array output and thrust vector during the long cruise period are shown in this section. Thruster sequencing and usage is shown diagrammatically in Fig. II-C-34.

Two communications blackouts occur when the sun-earth-space vehicle angle becomes less than five degrees. These are centered around 320 and 750 days from launch.

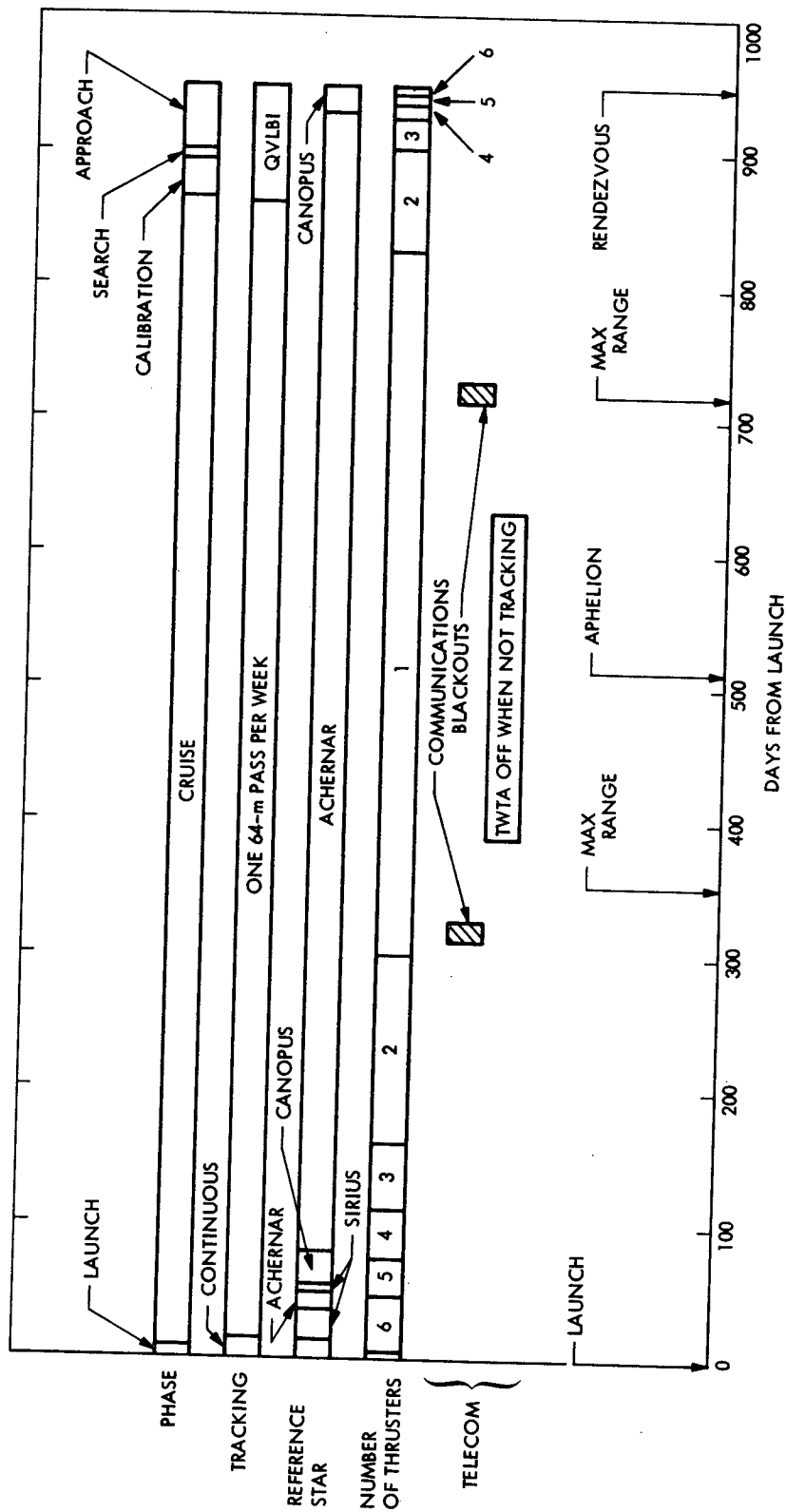
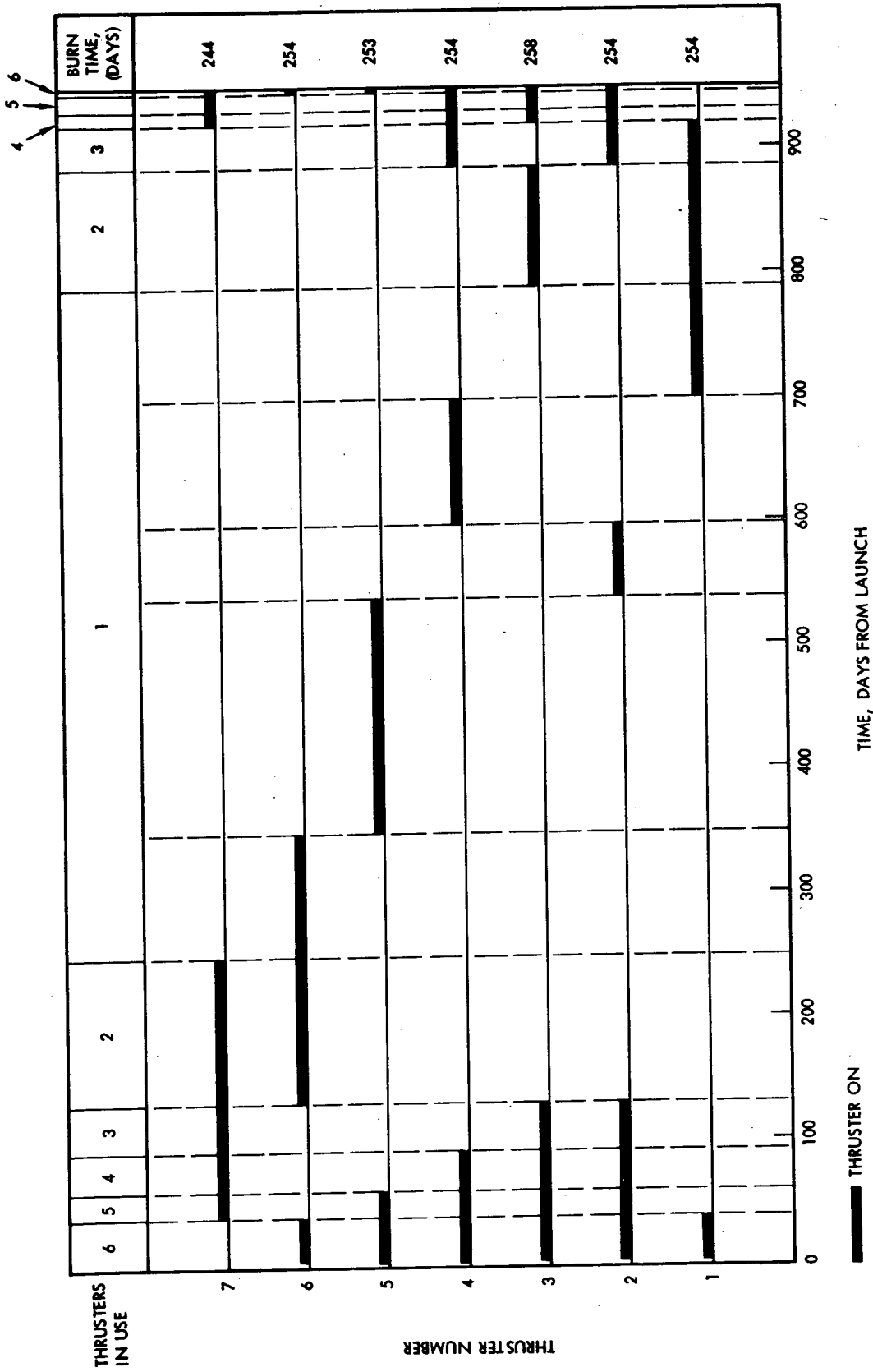


Fig. II-C-33. Major Mission Events Encke '80 Rendezvous Space Vehicle



NOTE: THIS BURN SEQUENCE IS PRODUCED TO MATCH THE POWER PROFILE GIVEN IN FIGURE II-C-4, USING A 2.6 KW, SEVEN-THRUSTER SYSTEM. THE DEVIATION FROM EQUAL BURN-TIME IS CAUSED BY THE PARTICULAR WAYS IN WHICH THE MISSION PHASES WERE DIVIDED.

Fig. II-C-34. Thruster Sequencing for an Equal-Burn Policy

c. Calibration Phase

A calibration phase permits the TV camera and scan control subsystem to be calibrated together using known star clusters for the approach guidance activities to follow. In-space calibration allows the gravity effects present during ground testing to be removed, significantly improving the pointing knowledge of the instruments. At this time, 8 kbps telecom performance becomes available.

To improve orbit determination accuracy at this time, very long baseline interferometer (QVLBI) tracking with two 64-m ground stations begins in this phase and continues until rendezvous.

d. Search Phase

In the Search Phase, pictures of Encke are attempted while maintaining a trajectory representing the best guess from earth-based data.

e. Approach Phase

Once the comet is acquired by the spacecraft TV camera, the approach guidance system derives the information necessary to ascertain the true cometary trajectory and the resultant changes to the space vehicle trajectory to effect a rendezvous.

Pictures of the comet's position are made daily and thrust vector changes are made as needed. A 16 kbps data rate can be sustained in this phase for visual imaging data. Rendezvous occurs when the space vehicle is within 1000 km of the nucleus at a relative rate of 4 m/sec or less. This should normally take place at  $L + 950$  days.

SEQUENCE OF EVENTS

LAUNCH PHASE

L = 0	Liftoff. Launch date: March 16, 1978
L + ___ min.	Achievement of transplanetary orbit
S = 0	Separation from launch vehicle
	Turn off tape recorder.
S + 2 min.	Activate A/C reaction gas system; begin sun search.
S + 30 min.	Acquire sun. Begin solar array deployment.
S + 40 min.	Solar Array deployed; begin battery charging.
S + 1 hr.	Deploy high-gain antenna.
L + 2 hr.	Turn on star tracker; begin search for star reference.
L + 5 hr.	Acquire star reference; gyros off.
• • • •	
L + 2 days	Activate thrust subsystem with preprogrammed thrust vector from computer. Six thrusters on. Continuous tracking for 11 days for intensive evaluation of thrust subsystem performance and accumulation of tracking data while in a thrusting mode.

CRUISE PHASE

- |             |  |
|-------------|--|
| L + 10 days | 1. Update and verify computer.         |
| L + 11 days | 1. Acquire Sirius with star tracker.   |
|             | 2. Change thrust vector.               |
|             | 3. Begin no-track intervals*.          |
| L + 18 days | 1. Update computer.                    |
|             | 2. Change thrust vector.               |
| L + 25 days | 1. Update computer.                    |
|             | 2. Change thrust vector.               |
| L + 32 days | 1. Update computer.                    |
|             | 2. Change thrust vector.               |
| L + 38 days | 1. Update computer.                    |
|             | 2. Acquire Achernar with star tracker. |
|             | 3. Change thrust vector.               |
|             | 4. Begin thrusting with 5 thrusters.   |
| L + 42 days | 1. Update computer.                    |
|             | 2. Change thrust vector.               |

---

\*Tracking only performed on days indicated. Each track begins with pointing the HGA towards earth.



CRUISE PHASE - continued

- |             |   |
|-------------|---|
| L + 47 days | <ol style="list-style-type: none"><li>1. Update computer.</li><li>2. Acquire Sirius with star tracker.</li><li>3. Change thrust vector.</li></ol>   |
| L + 51 days | <ol style="list-style-type: none"><li>1. Update computer.</li><li>2. Acquire Canopus with star tracker.</li><li>3. Change thrust vector.</li></ol>  |
| L + 58 days | <ol style="list-style-type: none"><li>1. Update computer.</li><li>2. Change thrust vector.</li></ol>  |
| L + 62 days | <ol style="list-style-type: none"><li>1. Update computer.</li><li>2. Change thrust vector.</li><li>3. Begin thrusting with 4 thrusters.</li></ol>   |
| L + 67 days | <ol style="list-style-type: none"><li>1. Update computer.</li><li>2. Change thrust vector.</li></ol>  |
| L + 72 days | <ol style="list-style-type: none"><li>1. Update computer.</li><li>2. Change thrust vector.</li></ol>  |
| L + 78 days | <ol style="list-style-type: none"><li>1. Update computer.</li><li>2. Acquire Achernar with star tracker.</li><li>3. Change thrust vector.</li></ol> |

CRUISE PHASE - continued

- |              |                                      |
|--------------|--------------------------------------|
| L + 85 days  | 1. Update computer.                  |
|              | 2. Change thrust vector.             |
| L + 92 days  | 1. Update computer.                  |
|              | 2. Change thrust vector.             |
|              | 3. Begin thrusting with 3 thrusters. |
| L + 99 days  | 1. Update computer.                  |
|              | 2. Change thrust vector.             |
| L + 106 days | 1. Update computer.                  |
|              | 2. Change thrust vector.             |
| L + 113 days | 1. Update computer.                  |
|              | 2. Change thrust vector.             |
| L + 120 days | 1. Update computer.                  |
|              | 2. Change thrust vector.             |
| L + 127 days | 1. Update computer.                  |
|              | 2. Change thrust vector.             |
| L + 132 days | 1. Update computer.                  |
|              | 2. Change thrust vector.             |
|              | 3. Begin thrusting with 2 thrusters. |

L + 139 days and thereafter on 7-day centers

1. Update computer.
2. Change thrust vector.

CRUISE PHASE - continued

L + 250 days      Begin thrusting with 1 thruster (No. 6).

•  
•  
•

L + 320 days      Communication blackout: Superior Conjunction.

•  
•  
•

L + 348 days      Continue thrusting with 1 thruster (No. 5).

•  
•  
•

L + 388 days      Begin period with TWTA off except on days of track:

Typical  
Sequence

1. Reduce thrust power level.
2. Turn on and warm up TWTA (2 hr.).
3. Update computer (2 hr).
4. Change thrust vector (4 hr.).
5. Turn off TWTA.
6. Resume increased thrust power level.

L + 508 days      Aphelion

•  
•  
•

L + 539 days      Continue thrusting with 1 thruster (No. 2).

•  
•  
•

CRUISE PHASE - continued

L + 599 days	Continue thrusting with 1 thruster (No. 4).
•	
•	
•	
L + 628 days	Resume TWTA on continuously.
•	
•	
•	
L + 700 days	Continue thrusting with 1 thruster (No. 1).
•	
•	
•	
L + 720 days	Communication Blackout: Superior Conjunction.
•	
•	
•	
L + 790 days	Begin thrusting with 2 thrusters.
•	
•	
•	
L + 868 days	Begin calibration phase.

CALIBRATION PHASE\*

L + 868 days      R - 82 days

1. Point HGA towards earth.
2. Turn on housekeeping science data.
3. Turn on A/G camera and scan control.  
Allow for warmup.
4. Point scan platform to star cluster.
5. Record series of pictures of different  
star clusters at various platform  
positions.
6. Turn off A/G camera and housekeeping  
science data.
7. Play back tape recorded pictures at  
8 kbps.

L + 875 days      R - 73 days

1. Point HGA towards earth.
2. Turn on housekeeping science data.
3. Turn on A/G camera and scan control.  
Allow for warmup.
4. Point scan platform to star cluster.
5. Record series of pictures of different  
star clusters at various platform  
positions.
6. Turn off A/G camera and housekeeping  
science data.
7. Play back tape recorded pictures at  
8 kbps.

\*Beginning with the calibration phase, daily tracking using QVLBI with a minimum of two 64-m ground stations will be used to improve OD accuracy.

CALIBRATION PHASE - continued

- |              |             |  |
|--------------|-------------|--|
| L + 882 days | R - 68 days | 1. Point HGA towards earth.  |
|              |             | 2. Turn on housekeeping science data.  |
|              |             | 3. Turn on A/G camera and scan control.<br>Allow for warmup.                                 |
|              |             | 4. Point scan platform to star cluster.  |
|              |             | 5. Record series of pictures of different<br>star clusters at various platform<br>positions. |
|              |             | 6. Turn off A/G camera and housekeeping<br>science data.                                     |
|              |             | 7. Play back tape recorded pictures at<br>8 kbps.  |

L + 886 days	R - 64 days	Begin thrusting with 3 thrusters.
--------------	-------------	-----------------------------------

L + 890 days	R - 60 days	Begin search phase.
--------------	-------------	---------------------

SEARCH PHASE

- |              |             |  |
|--------------|-------------|--|
| L + 890 days | R - 60 days | 1. Point HGA towards earth.  |
|              |             | 2. Turn on A/G Camera and housekeeping data. Allow for warmup.               |
|              |             | 3. Point scan platform in direction most likely to acquire Encke.            |
|              |             | 4. Fill tape with pictures starting at this position and spiralling outward. |
|              |             | 5. Play back entire tape at 8 kbps.  |
| L + 891 days | R - 59 days | Repeat (R - 60 days).  |
| L + 892 days | R - 58 days | Repeat (R - 60 days).  |
| L + 893 days | R - 57 days | Repeat (R - 60 days).  |
| L + 894 days | R - 56 days | Repeat (R - 60 days).  |
| L + 895 days | R - 55 days | Repeat (R - 60 days).  |
|              |             | 6. Update computer.  |
|              |             | 7. Change thrust vector.   |
|              |             | 8. Expected acquisition.   |
| L + 896 days | R - 54 days | Begin approach phase ( R - 50 days at latest).                               |

APPROACH PHASE

L + 900 days      R - 50 days

1. Change HGA pointing direction.
2. Update computer.
3. Change thrust vector.
4. Record full tape load of pictures at 16 kbps.

L + 901 days      R - 49 days

Repeat above daily until Rendezvous, changing thrust vector, number of thrusters, HGA pointing, and reference star, as needed. QVLBI tracking used with 2 ground stations participating.

L + 918 days      R - 32 days

Begin thrusting with 4 thrusters.

L + 927 days      R - 23 days

1. Update computer.
2. Terminate thrusting.
3. Acquire Canopus with star tracker.
4. Change thrust vector.
5. Resume thrusting.

L + 932 days      R - 18 days

Begin thrusting with 5 thrusters.

L + 948 days      R - 2 days

Begin thrusting with 6 thrusters.

L + 950 days      R = 0

Rendezvous.



## D. SPACE VEHICLE SYSTEM DESCRIPTION

### 1. Configuration

The space vehicle configuration concept shown in Figs. II-D-1 and II-D-2 evolved after several iterations of a basic modular assembly approach utilizing open truss structure and Viking technology.

The space vehicle may be divided into two modules: the spacecraft module and the SEP module. The SEP module is composed of two major assemblies: the SEP module support subsystems and the thrust subsystem. These are shown in Fig. II-D-2. All three units are open box-like structures which serve as the primary structure and provide support for the subsystem equipment mounted thereon. A more detailed description of the space vehicle's structure may be found in Sections E. 7 and F. 4.

An isometric view of the space vehicle shown in Fig. II-D-2 is exploded to depict and emphasize the modular approach to the vehicle's construction, whereas Fig. II-D-1 is a plan view of the space vehicle in a flight mode as well as in a launch configuration.

#### a. Spacecraft

By viewing Fig. II-D-2, it may be seen that the spacecraft may be separated from the SEP module at the four corners of the interconnecting truss structure. The spacecraft carries most of the basic equipment and subsystems found on interplanetary spacecraft. The box structure supports eight bays of electronic assemblies and the required interconnecting harnesses and cables. Mounted to the forward end of the primary structure is a 1.47 m (58 in.) diameter high-gain antenna, whose two degrees of freedom enables it to point at earth during various phases of the Encke Rendezvous mission. As shown in the launch mode views of Fig. II-D-1, the high-gain antenna is stowed and tied down to the support structure during launch. After launch, tiedown release devices will be actuated to enable the antenna to rotate about its two axes of rotation.

DO NOT PHOTOGRAPH

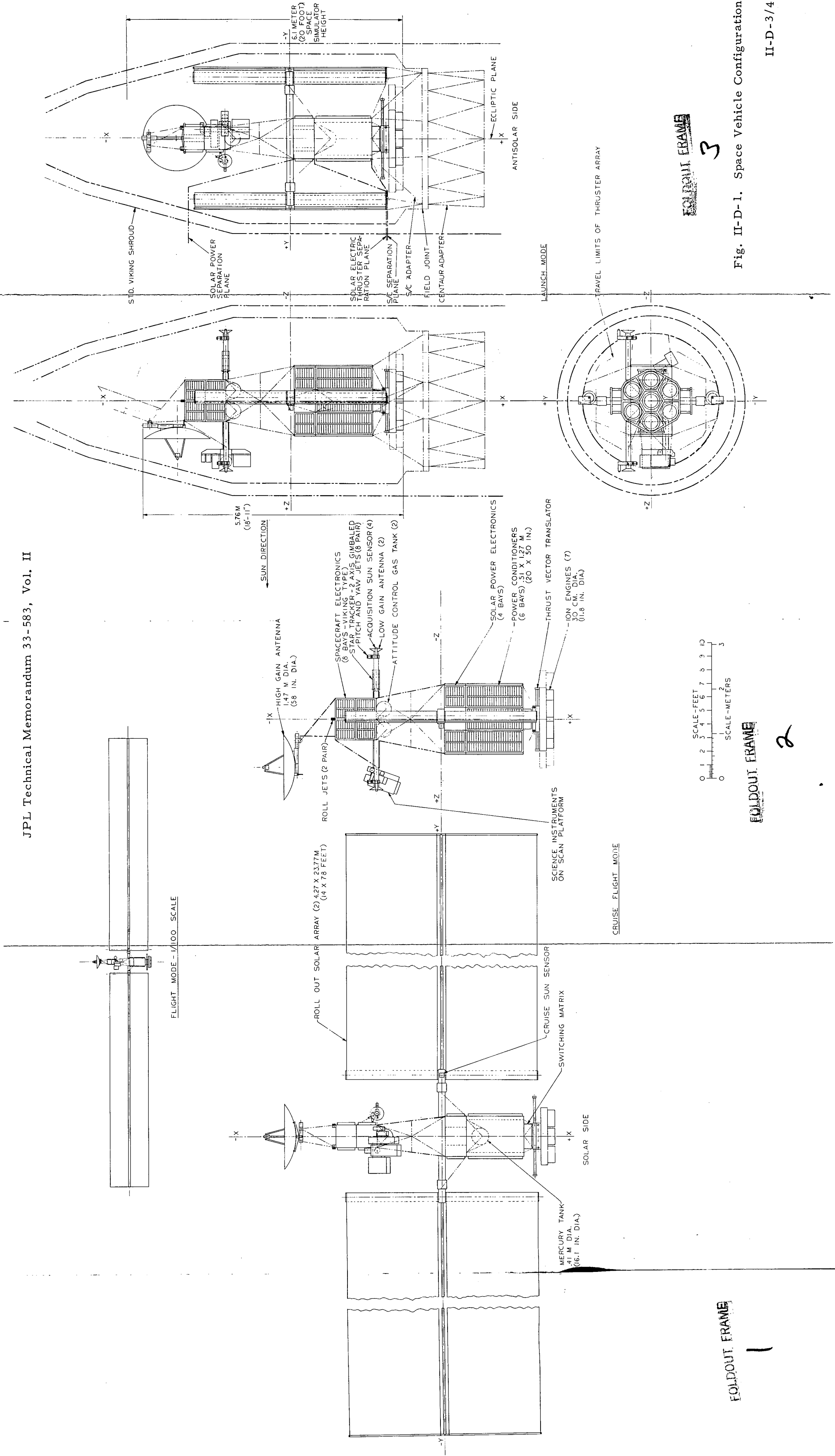


Fig. II-D-1. Space Vehicle Configuration

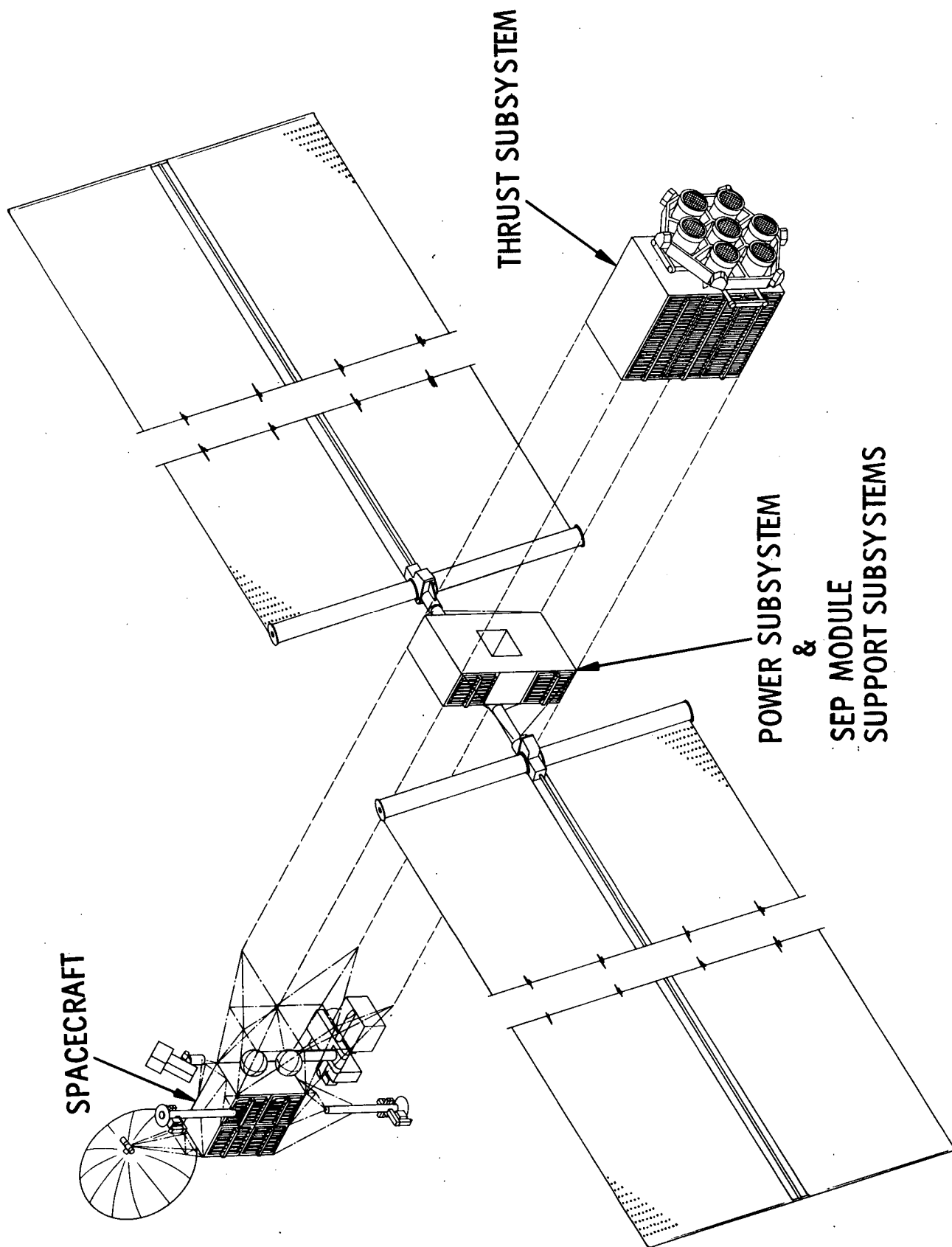


Fig. II-D-2. Space Vehicle (Exploded View)

Referring to Fig. II-D-1, two low-gain antennas are mounted to the spacecraft primary structure on the sides of the box apposite the louvered electronic bays. Using two low-gain antennas on opposite sides of the spacecraft provides  $4\pi$  steradians of coverage for telecommunications. This figure shows the star tracker mounted on the spacecraft's anti-sun side (end view of launch configuration) and is two-axis-gimballed. The gimbaling is necessary to enable the instrument to lock on to several different stars during the Encke mission for roll attitude stabilization without stray light interference from the solar arrays, or other external equipment.

In this figure, a science scan platform is depicted on the side opposite the star tracker. It has two-degrees-of-freedom and its height above structure and the ion engine array has been set to accommodate viewing the comet from 60 days prior to, and during rendezvous.

Attitude control gas tanks nested to the underside of the spacecraft are shown in Fig. II-D-1. These tanks supply gas to the roll jets mounted to the top side of the box structure near the high-gain antenna and pitch and yaw jets mounted on the low-gain antenna masts. Acquisition sun sensors are also mounted on these masts.

The center assembly previously denoted as the SEP module support subsystem is more easily described by using Fig. II-D-2. This isometric drawing shows the module to be made up of four electronic bays housing the power subsystem and associated electronics in a box structure. Mounted to the box structure are tubular supports that carry each  $0.27 \times 23.77$  m (10.6 in.  $\times$  78 ft) rollout solar array, its deployment actuator, and orientation drive mechanisms. Fig. II-D-1 shows a cruise sun sensor mounted on the sun side of the deployment actuator.

The third assembly shown in Fig. II-D-2 is the thrust subsystem. This subsystem is made up of six power conditioners and seven 30-cm ion engines arranged in a hexagonal array. The thrust vector translator, switching matrix, and mercury propellant tank are shown in the cruise flight mode, solar side area of Fig. II-D-1. All of the above-mentioned equipment are mounted to

the primary structure. The power conditioners are mounted to the sides of the structure so that their louvered sides lie in a plane parallel to the sun's rays during flight. This arrangement observes the temperature control requirement of the power conditioners, that they should never be subjected to direct solar energy, and allow direct radiation to black space. The ion engine array is attached to the box structure of the power conditioners and supported by tubular truss members. As shown in the cruise flight mode area of Fig. II-D-1, the switching matrix is nested between the power conditioner structure and the ion engine array to maintain minimum cable length. The mercury propellant tank will be supported by truss work within the box structure of the power conditioners.

b. Shuttle-Space Vehicle Adaptation Concept

The space vehicle is shown in Fig. II-D-3 in its stowed configuration inside the shuttle bay. Two trusses support the vehicle during launch and flight operations. One truss, engine mount type, adapts the vehicle to the Centaur booster. The second is a "W" type of truss and supports the space vehicle near the ion thruster array.

c. Packaging Arrangement

The electronic equipment packaging arrangement is based on use of standard Viking '75 electronic chassis and subassemblies located in two separate compartments - the spacecraft and the SEP module.

The spacecraft compartment (Fig. II-D-4) contains eight electronic assemblies, five of which consist of identical or slightly modified Viking equipment. The other three assemblies used, consist primarily of Viking subsystem electronics and include the TV electronics. Approximately 25 percent of the volume of the three assemblies is available for subsystem change and to comply with science electronic requirements.

The equipment is located to maintain subsystems within an assembly, provide the shortest rf cable to the antenna, distribute the power dissipation within the compartment, minimize power cable losses and group the signal and logic cables.

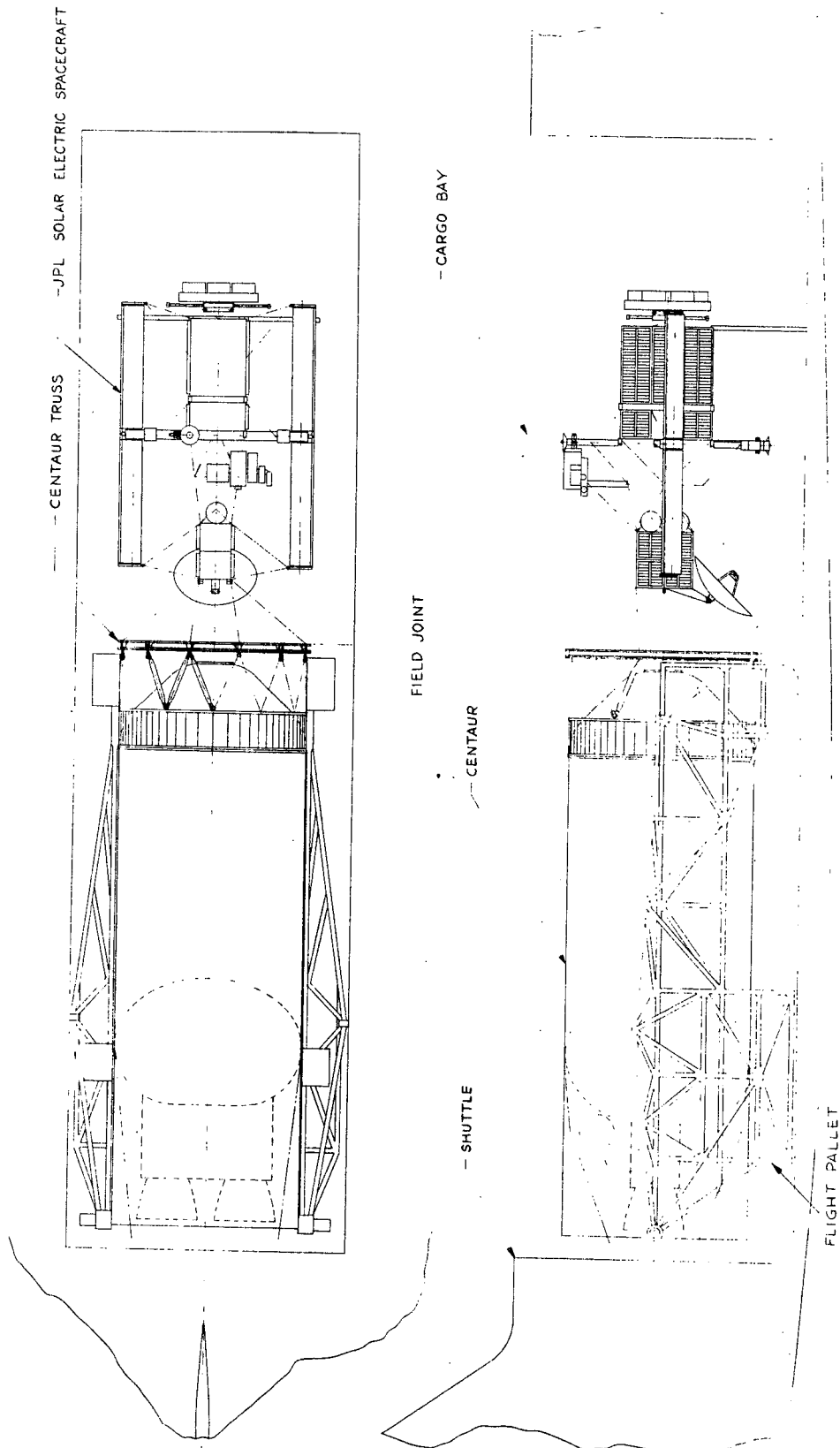


Fig. II-D-3. Space Vehicle Stowed in Shuttle Bay

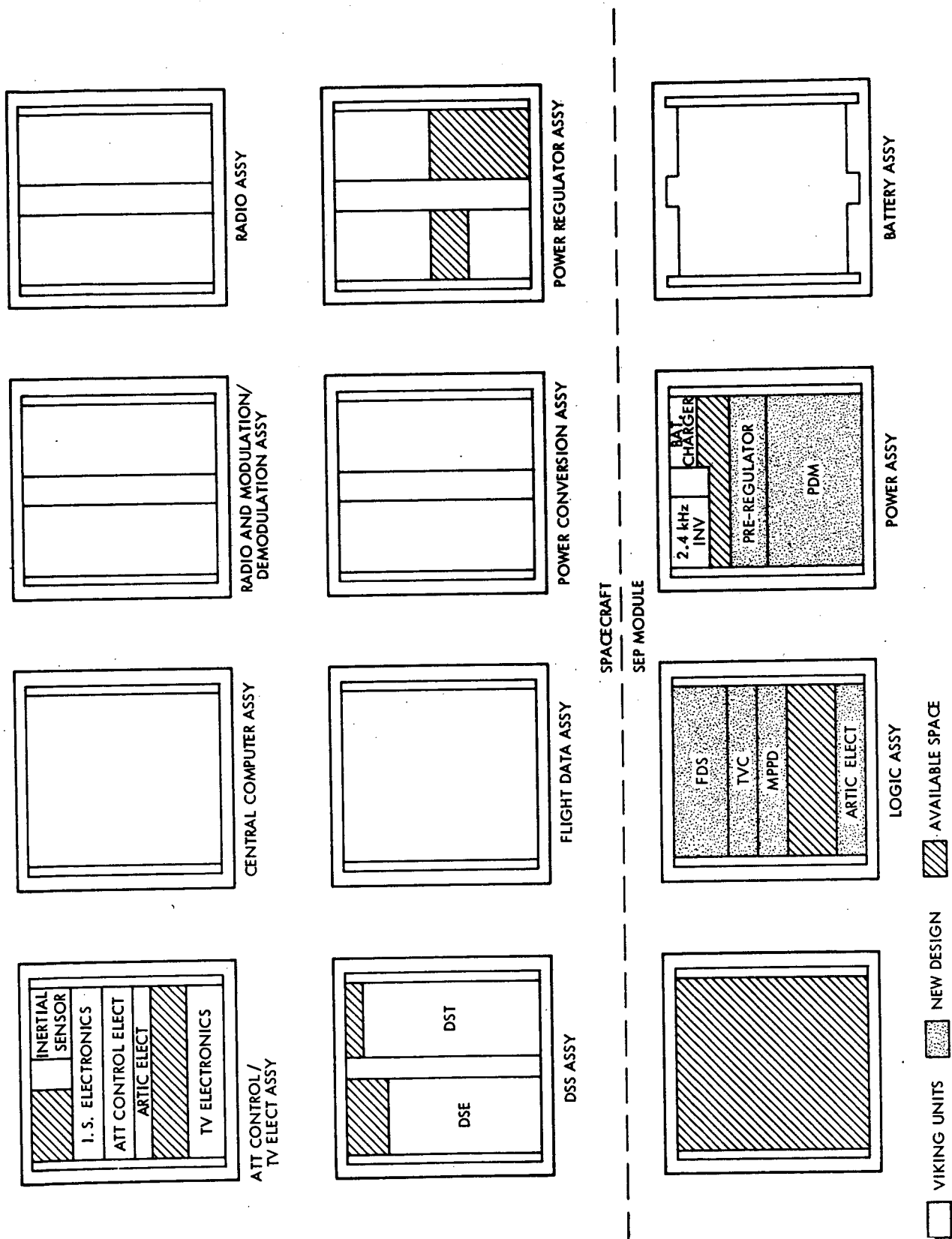


Fig. II-D-4. Spacecraft Compartment Packaging Arrangement



The four-bay SEP module compartment contains the electronic assemblies complying with the solar electric propulsion requirements, including the power distribution module located for minimum system power loss. One bay of the SEP module is available for additional electronics, if required, with 18 percent of the space in the other two electronics bays available. The Viking battery takes a full bay.

## 2. Equipment List, Weights and Spacecraft Mass Properties

The weights shown in Tables II-D-1 and II-D-2 are preliminary and are presented by subsystem breakdown of the spacecraft and SEP module. The tables highlight the following.

	<u>kg</u>	<u>lb</u>
Spacecraft weight:	448.1	985.8
SEP module weight:	814.1	1791.0
Launch vehicle adapter weight:	27.2	59.8
Propellant weight:	480.0	1056.0

A preliminary analysis of the mass distribution in the vehicle locates its center of gravity for the flight configuration described in the column headings of Table II-D-2. The moments of inertia of the vehicle have been calculated as shown in Table II-D-2 for the individual flight configuration denoted by the column heading.

Table II-D-1. Equipment List and Mass Allocations

Item	Mass		Item	Mass	
	kg	lb		kg	lb
<b>SPACECRAFT</b>			<b>SEP MODULE</b>		
<u>Science</u>			<u>Thrust</u>		
*TV (100 $\mu$ rad res.)	17.4	38.3	Thrusters (7)	51.0	112.2
*IRR	1.9	4.2	Power Conditioners (6)	98.0	215.6
*White Light Photometer	2.5	5.5	TVC Mechanism	39.7	87.2
*Photopolarimeter	4.2	9.2	Propellant Tankage (1)	15.0	33.0
*Mass Spectrometer	4.5	9.9	Switching Matrix (1)	12.3	27.1
*Microwave Altimeter	6.0	13.2	Cabling	8.4	18.5
*Radiometer (UV, 1000-4500 Å)	2.5	5.5	Contingency	10.0	22.0
*Plasma Probe	1.9	4.2		234.4	515.6
*Mass Spectrometer	4.5	9.9	<u>Power</u>		
Optical Particle Detector (Sisyphus)	3.0	6.6	Solar Arrays (2)	315.0	693.0
Magnetometer	2.2	4.9	Battery	30.5	67.1
Plasma Wave Detector	4.5	9.9	Battery Charger	1.5	3.3
Langmuir Probe	1.9	4.2	Preregulator	3.5	7.7
	57.0	125.4	Power Distribution	13.6	29.9
<u>Computer</u>	25.0	55.0	2.4 kHz Inverter	1.8	4.0
<u>Flight Data</u>	20.0	44.0	Maximum Power Point Detector	4.5	9.9
<u>Data Storage</u>	28.8	63.3		370.4	814.9
<u>Telecommunications</u>			<u>Thrust Vector Control</u>		
RFS	33.5	73.7	TVC Electronics	2.7	5.9
MDS	8.4	18.5	Cruise Sun Sensor	0.2	0.4
Low-Gain Antenna (2)	3.4	7.5	Translator Actuators	5.6	12.0
High-Gain Antenna (1)	5.9	13.0	Gimbal Actuators	10.9	24.0
X-Band Transmitter	1.1	2.4	Solar Array Rotators (2)	18.1	39.8
	52.3	115.1		37.5	82.5
<u>Mechanical Devices</u>			<u>Flight Data</u>		
HGA Articulation	1.0	2.2	Master FDS	9.1	20.0
HGA Latches	3.4	7.5	FDS Slave Allowance	5.0	11.0
Scan Platform Latches	6.8	15.0		14.1	31.0
Star Tracker Articulation	0.5	1.1	<u>Mechanical Devices</u>		
Star Tracker Latch	0.9	2.0	Solar Array Gimbal Latch (2)	9.1	20.0
Staging Latch	4.5	9.9	Thruster Array Latch (4)	3.6	7.9
Bus Louvers (8)	5.4	11.9	Electronic Bay Louvers (4)	2.7	5.9
	22.5	49.6	PC Louvers (5)	16.3	35.9
<u>Thermal Control</u>				31.7	69.7
Bus Thermal Blankets	1.9	4.2	<u>Cabling</u>		
Scan Platform Thermal Blankets	1.4	3.1	Power Cabling	11.5	25.3
<u>Attitude Control</u>	3.3	7.3	Signal Cabling	10.0	22.0
Attitude Control Electronics (1)	3.7	8.1		21.5	47.3
Inertial Electronics (1)	2.7	5.9	<u>Structure</u>		
Inertial Sensors (1)	2.3	5.1	Primary Truss	24.6	54.1
Sun Gate (1)	0.1	0.2	Solar Array Support	6.4	14.1
Acquisition Sun Sensors (4)	0.2	0.4	Power Conditioner Frame	31.8	70.0
Star Tracker (1)	4.2	9.2	Propellant Tank Support	11.4	25.1
N <sub>2</sub> Tanks (2)	10.0	22.0	Switching Matrix Chassis (1)	2.2	4.8
High-Pressure Module (2)	4.7	10.3	Electronics Chassis (4)	11.0	24.2
Low-Pressure Module (4)	2.6	5.7	Meteoroid Protection	8.2	18.0
Thruster Assemblies	1.2	2.6	Cabling Troughs	2.0	4.4
N <sub>2</sub>	14.1	31.0		97.6	214.7
Articulation Control Electronics (1)	4.0	8.8	<u>Thermal Control</u>		
High-Gain Antenna Actuators (2)	2.3	5.1	PC Thermal Blankets	3.9	8.6
Scan Control Actuators (2)	4.3	9.5	Bus Thermal Blankets	2.0	4.4
Star Tracker Actuators (2)	1.0	2.2	Thruster Array Thermal Blankets	1.0	2.2
	57.4	126.1		6.9	15.2
<u>Cabling</u>			<b>Total SEP Module</b>	<b>814.1</b>	<b>1791.0</b>
Bus Cabling	55.8	122.8	<b>LAUNCH VEHICLE ADAPTER</b>		
Scan Platform Cabling	4.2	9.2	Structure	20.4	44.9
<u>Power</u>			Release Mechanisms	2.5	5.5
Power Sourced Logic	7.1	15.6	Cabling	2.5	5.5
Booster Regulator A	3.5	7.7	Solar Array End Latches	1.8	4.0
Booster Regulator B	3.5	7.7		27.2	59.8
Power Control	2.5	5.5	<b>SPACE VEHICLE SUMMARY</b>		
Power Distribution	4.5	9.9	Spacecraft	448.1	985.8
2.4 kHz Main Inverter	2.1	4.6	SEP Module	814.1	1791.0
2.4 kHz Standby Inverter	2.1	4.6	Launch Vehicle Adapter	27.2	59.8
400 Hz Inverter	1.8	4.0	Propellant	480.0	1056.0
	27.1	59.6	<b>Launch Gross</b>	<b>1769.4</b>	<b>3892.7</b>
<u>Structure</u>					
Primary Truss	32.0	70.4			
Bus	18.0	39.6			
Electronic Chassis (8)	22.0	48.4			
Bus Cable Trough	1.2	2.6			
Attitude Control Tank Support	1.1	2.4			
Scan Platform Frame and Boom	15.9	35.0			
Bus Meteoroid Protection	4.5	9.9			
	94.7	208.3			
<b>Total Spacecraft</b>	<b>448.1</b>	<b>985.8</b>			

Table II-D-2. CG and Inertial Properties

Mission Mode	Solar Array Deployed			
	Solar Array Stowed	Launch + 1 Hour Drums Parallel to X-Axis	Launch + 41 Days Drums Perpendicular to X-Axis	Launch + 500 Days Drums Parallel to X-Axis
Propellant on Board (kg)	480	480	430	190
Center of Gravity mm (in)				
X c.g.	221 (8.7)	224 (8.9)	196 (7.7)	97 (3.8)
Y c.g.	-8 (-.3)	-13 (-.5)	-13 (-.5)	-15 (-.6)
Z c.g.	38 (1.5)	51 (2.0)	53 (2.1)	61 (2.4)
Moment of Inertia $\text{km}^2$ (slug $\text{ft}^2$ )				
Ix	908 (670)	48,356 (35,666)	48,769 (35,971)	48,722 (35,936)
Iy	3347 (2469)	3,250 (2,397)	3,235 (2,386)	3,070 (2,264)
Iz	3822 (2819)	51,180 (37,749)	50,750 (37,432)	50,622 (37,337)
Product of Inertia $\text{km}^2$ (slug $\text{ft}^2$ )				
Ixy	20 (15)	31 (23)	31 (23)	23 (17)
Iyz	-15 (-11)	-26 (-19)	-24 (-18)	-28 (-21)
Ixz	-150 (-111)	-144 (-106)	-142 (-105)	-4,299 (-3,171)

3. Space Vehicle System Functional Block Diagram

The functional block diagram of the space vehicle system is shown in Fig. II-D-5. The interfaces between the SEP module and spacecraft are shown to be:

- (1) SEP module master flight data subsystem to spacecraft flight data subsystem - whereby SEP module telemetry is routed to the spacecraft data handling system for storage and/or transmission to earth.
- (2) SEP module master flight data subsystem to and from spacecraft computer command subsystem - whereby SEP module is supplied to the spacecraft computer and computer-generated and/or ground commands are routed to the SEP module.
- (3) SEP module battery to spacecraft power subsystem - whereby battery power is supplied to the spacecraft.
- (4) SEP module power preregulator to spacecraft power subsystem - whereby solar array power is supplied to the spacecraft.
- (5) Spacecraft celestial sensors to SEP module TVC electronics - whereby attitude references are supplied to the SEP module TVC.

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4. Data Handling and Commands

a. Data Handling

The spacecraft will contain equipment to gather information about the performance and status of spacecraft engineering subsystems and science instruments, about the results of these instruments; and about the status of the SEP module subsystems. The information gathered will be processed into specific formats and either stored or presented directly to on-board telecommunications equipment for transmission to earth. All data handling and processing will be accomplished by the flight data subsystem (FDS), data storage subsystem (DSS), computer command subsystem (CCS) and modulation/demodulation subsystem (MDS).

The SEP module will contain equipment to gather information on the status of SEP module subsystems. The information gathered will be processed into specific formats and either delivered to the spacecraft FDS and/or to the CCS. The data handling in the SEP module will be accomplished by the SEP module master flight data subsystem (MFDS). A data handling schematic is shown in Fig. II-D-6.

b. Commands

Commands are required to carry out flight sequences and counter unexpected events. These commands may be issued in a predetermined timing sequence via on-board program control or as received from the ground. Commands are received from earth via the S/X-band antenna subsystem (SXA) by the radio frequency subsystem (RFS), demodulated by the modulation/demodulation subsystem (MDS), and decoded by the CCS. The CCS routes commands to the spacecraft subsystems and the SEP module MFDS.

Elements of the SXA, RFS, MDS, CCS, and MFDS comprise the command system. Fig. II-D-7 shows the functional organization of these elements, whose individual roles are

- (1) SXA - Receives the uplink S-band carrier containing the command modulation, with either antenna, and routes it to RFS via coaxial cables.
- (2) RFS - Demodulates the command subcarrier from the uplink S-band signal, and sends it to an MDS command detector unit (CDU). Only one RFS receiver and its corresponding MDS CDU is operating at a time.
- (3) MDS - Establishes an in-lock condition for the command detector, detects the command data bits in the command subcarrier and sends them to both CCS processors.
- (4) CCS - Decodes the data bits to detect valid commands, which are executed ten bit-times after receipt if the CDU remains in lock, and distributes either stored or direct ground commands.
- (5) MFDS - Accepts and decodes commands from CCS and routes commands to user subsystem.



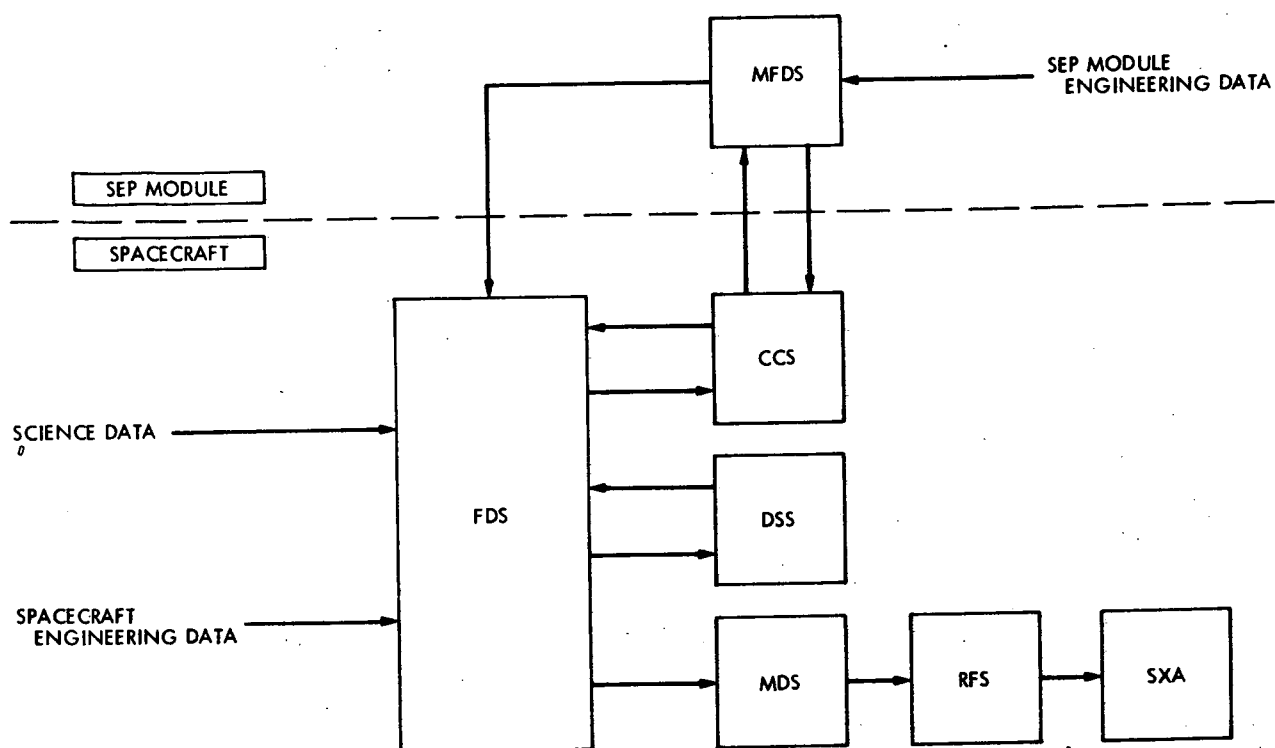


Fig. II-D-6. Data Handling System Functional Block Diagram

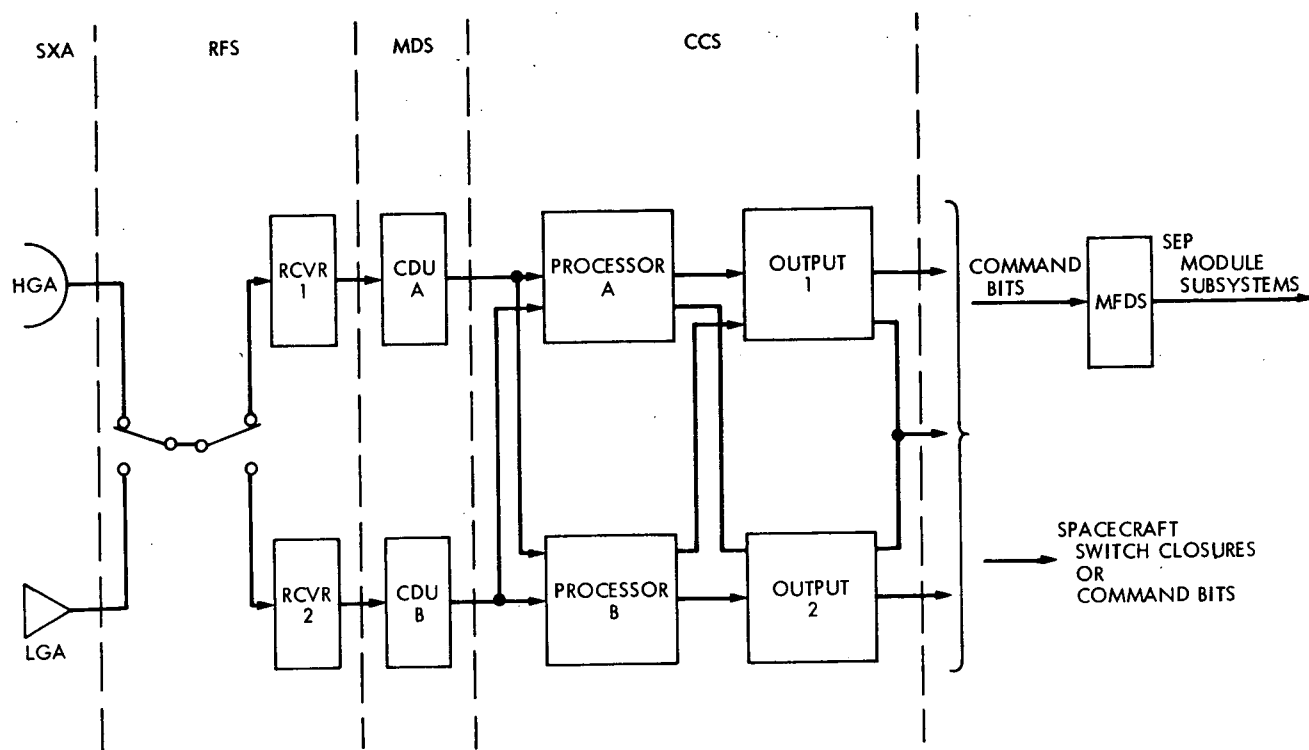


Fig. II-D-7. Command System Functional Block Diagram

5. Power Management

Power requirements for both the spacecraft and SEP module portion of the space vehicle are shown in Table II-D-3 along with a summary of the entire space vehicle requirements. The mission is subdivided into key phases which correspond to either launch activity (which is critical to assess battery requirements), or numbers of operating thrusters. The table shows a satisfactory power margin throughout the mission.

6. Electronic Packaging

a. Scope

1) Electronic Packaging Requirements. This section specifies the approved packaging techniques for flight electronic equipment. Packaging considerations relative to structural, electrical, thermal and configurational requirements are identified.

2) Applicability. The design techniques expressed herein are to be applied to the electrical packaging of flight electronic equipment. Where special requirements are recognized, calling for packaging techniques different from those defined in this document, the Electronic Packaging Engineer shall qualify and approve, in advance, the use of alternative techniques.

b. Functional Description

1) General Packaging Design Requirements.

- (a) Electronic equipment shall be designed as replaceable assemblies. Each assembly shall, in general, be comprised of replaceable sub-assemblies; assemblies and subassemblies shall be of standardized sizes and shapes to provide design flexibility. Where practical, electronic equipment will be contained in the bays of the SEP primary structure, however, scientific

Table II-D-3. Power Utilization Summary

Flight Modes	① Liftoff	② Sun Acquisition & Battery Charging	③ Elect Prop & Cruise Science (6 Thrust)	④ / ⑫ Elect Prop & Cruise Science (5 Thrust)	⑤ / ⑪ Elect Prop & Cruise Science (4 Thrust)	⑥ / ⑩ Elect Prop & Cruise Science (3 Thrust)	⑦ / ⑨ Elect Prop & Cruise Science (2 Thrust)	⑧ Elect Prop & Cruise Science (1 Thrust)	⑬ Encounter Elect Prop & Science (6 Thrust)
SPACE VEHICLE SUMMARY*									
Time - Days	1 hr.	2	36	24/16	29/14	41/32	118/96	540	2
Power Source	Battery	S/P	S/P	S/P	S/P	S/P	S/P	S/P	S/P
SEP Module & Spacecraft***	393	604	492	498	495	489	493	495 (425)***	659
Propulsion	0	0	15,700	13,150	10,520	7,890	5,260	2630/1615****	15,500
Dist. Losses	0	10	200	160	120	0	0	60/40	200
Total Unregulated Power Requirements	393	614	16,392	13,808	11,135	8,479	5,839	3115/2080***	16,359
Solar Array Output Capability		20,000	20,000	16,900	13,600	10,390	7,140	3890/2620	20,000
Output with 18% Degradation		16,400	16,400	13,850	11,150	8,500	5,850	3190/2100	16,400
SPACECRAFT									
A. 2.4 kHz Inverter Loads									
Science	0	0	17	17	17	17	17	17	72
Attitude Control	30.0	31.7	10	10	10	10	10	10	10
Data Storage	17	0	0	0	0	0	0	0	30
Flight Data	35	35	35	35	35	35	35	35	35
CCS	20	20	20	20	20	20	20	20	20
Mod/Demod	8.7	8.7	8.7	8.7	8.7	8.7	8.7	8.7	8.7
RFS	28	28	28	28	28	28	28	28	28
Gyro Elect	27.7	27.7	0	0	0	0	0	0	0
Pyro	0	1.0	1.0	1.0	1.0	1.0	1.0	1.0	1.0
Power Distr.	13	13	13	13	13	13	13	13	13
Scan Platform	0	0	0	0	0	0	0	0	35.3
2.4 kHz Output	199.5	165.1	132.7	132.7	132.7	132.7	132.7	132.7	253.0
2.4 kHz Inv. Input (eff. 90%)		183	147	147	147	147	147	147	281
B. 400 Hz Inverter Loads									
Gyros	18	18	0	0	0	0	0	0	0
Input (eff. 84%)	21.4	21.4	0	0	0	0	0	0	0
C. 30 Volt Conv. Loads									
Eng Valves	0	0	0	0	0	0	0	0	0
Gimbal	0	0	0	0	0	0	0	0	0
Booster Regulator Loads									
2.4 kHz Input	199.5	183	132.7	132.7	132.7	132.7	132.7	132.7	253
400 Hz Input	21.4	21.4	0	0	0	0	0	0	0
Heaters	60	60	60	60	60	60	60	60	60
Total Loads	280.9	264.4	182.7	182.7	182.7	182.7	182.7	182.7	313
Boost Reg (eff. 88%)	320	300	218	218	218	218	218	218	356
Unregulated Power									
Booster Input	320	300	218	218	218	218	218	218	256
Battery Charger	0	93	36	36	36	36	36	36	36
Power Failure Detector	1.5	1.5	1.5	1.5	1.5	1.5	1.5	1.5	1.5
TWT	59	59	59	59	59	59	59	59	59
XTX	0	0	0	12.7	0	0	0	0	12.7
Total Unregulated Power	380.5	453.5	314.5	327.2	314.5	314.5	314.5	314.5	465.2
Input to PS&L (eff. 97)	393	467	324	337	324	324	324	324 (263)***	376
Input to Spacecraft from SEP Module		467	324	337	324	324	324	324 (263)***	476
SEP MODULE									
A. 2.4 kHz Inverter Loads									
FDS	0	0	15	15	15	15	15	15	15
Prop. Power Cond.	0	0	15	12.5	12.5	10	50	25	150
Total	0	0	30	27.5	25	22.5	20	17.5	30
Inv. Input (eff. 9)	0	0	34.3	30.5	28	25	22.2	19.5	34.3
B. DC Regulator Loads									
2.4 kHz Inverter Input	0	0	34.3	30.5	28	25	22.2	19.5	34.3
TVC	0	0	68	54	54	40	40	26	68
Heaters for P. C. Units	0	72	0	12	24	36	48	60	0
Solar Array Pointing	0	15	15	15	15	15	15	15	15
Total Regulated Power	0	87	117.3	111.5	121	116	125.2	120.5	117.3
*Power in watts **Includes a preregulator efficiency of 0.9 ***TWT is off for minimum power mode ****Thruster at less than full thrust output									

instruments, sun sensors and other equipment with special viewing requirements will be mounted to satisfy those requirements.

- (b) Electronic equipment located in the bays shall be packaged in standardized electronic assemblies. The packaging techniques required for equipment located in bays shall also be specified for equipment located elsewhere consistent with structural, electrical, thermal and physical considerations peculiar to such equipment.
- (c) Electronic packaging design shall meet the requirements for survival and reliable operation in space. Adequate protection against degradation during bench handling, testing, shipping and storage shall be provided. Adequate access to equipment for adjustment, repair and modification shall also be provided
- (d) General electronic packaging design requirements are specified by TSS-220-1,\* Electronic Packaging. Refer to Appendix B.

2) System Requirements. Electronic packaging shall conform to the guidelines established in design criteria as would be defined in a flight project, and meet the requirements contained in Section B, Requirements and Constraints.

3) Electrical Requirements

(a) Requirements contained in VO75-3-260,\* Electrical Grounding and Interfacing, shall be met, relative to:

- (1) Chassis - circuitry isolation
- (2) Electrical bonding
- (3) Magnetic fields

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\*JPL internal documents.

- (4) Grounding of shields
- (5) Grounding of reference trees

(b) Equipment developing or using voltage greater than 250 V during any mission phase shall meet the JPL design and testing requirements specification, DM505139,\* High Voltage Protection, to prevent corona, or other forms of arcing, in atmospheric pressures ranging from sea level to the vacuum of space. This requirement applies even to equipment that is not powered during the ascent through the critical pressure region.

c. Electronic Assembly Design Requirements

1) Packaging Types. Electronic Assemblies shall employ, as appropriate, two standard packaging techniques:

- (a) Integral Shearplate Packaging
- (b) Plug-in Equipment Packaging

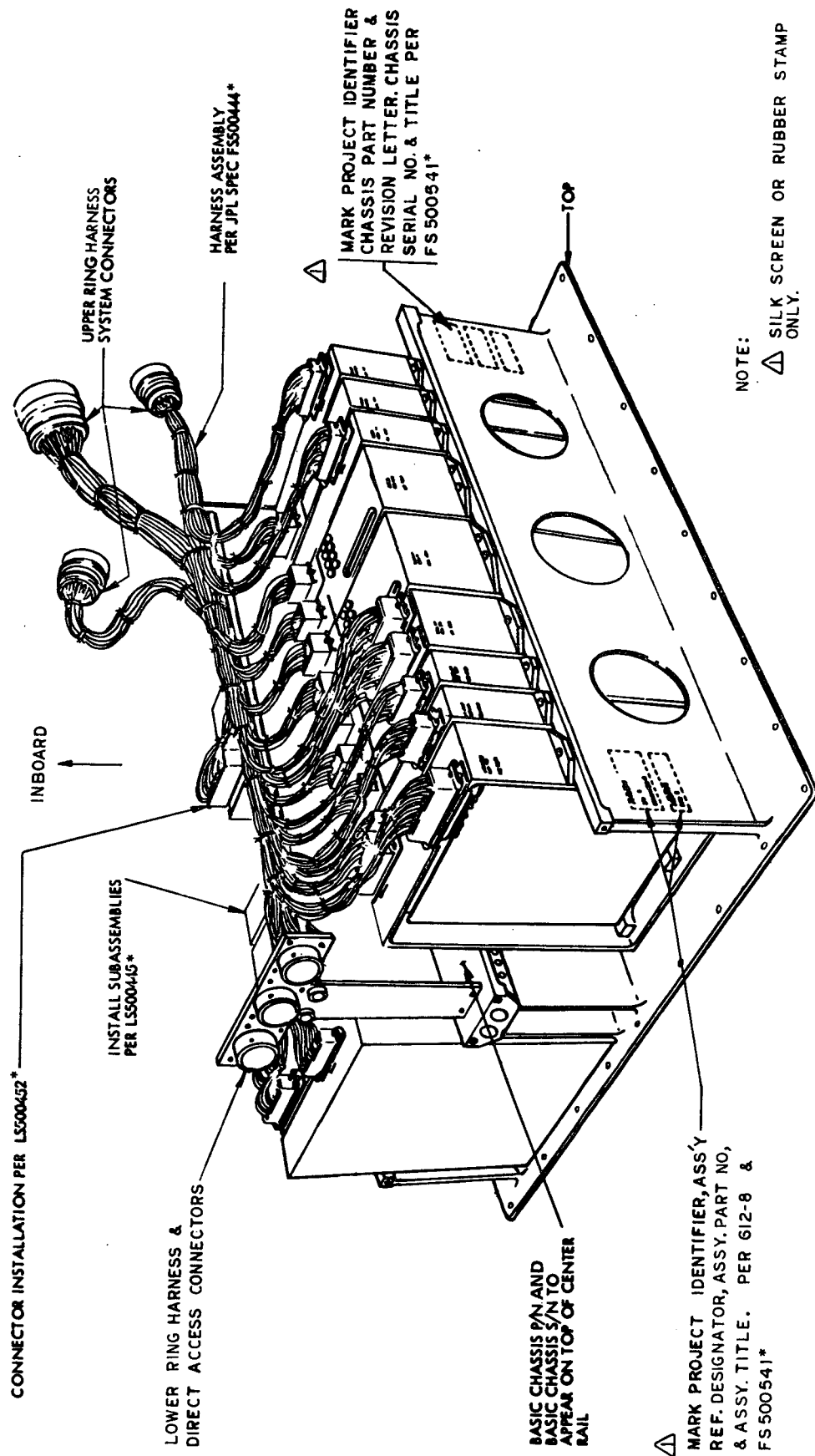
Integral shearplate electronic assemblies shall conform to the geometry indicated in Fig. II-D-8. Plug-in equipment electronic assemblies shall conform to the geometry indicated in Fig. II-D-9. The battery electronic assemblies shall be specially packaged.

2) Structural Requirements

- (a) The electronic assemblies mounted in the bays of the primary structure shall form an integral part of that structure. The electronic subassemblies contained within each electronic assembly contribute to the strength and rigidity of the assembly. These characteristics are derived principally from the chassis of the electronic assembly and from the subchassis of each electronic subassembly. Other structural requirements for

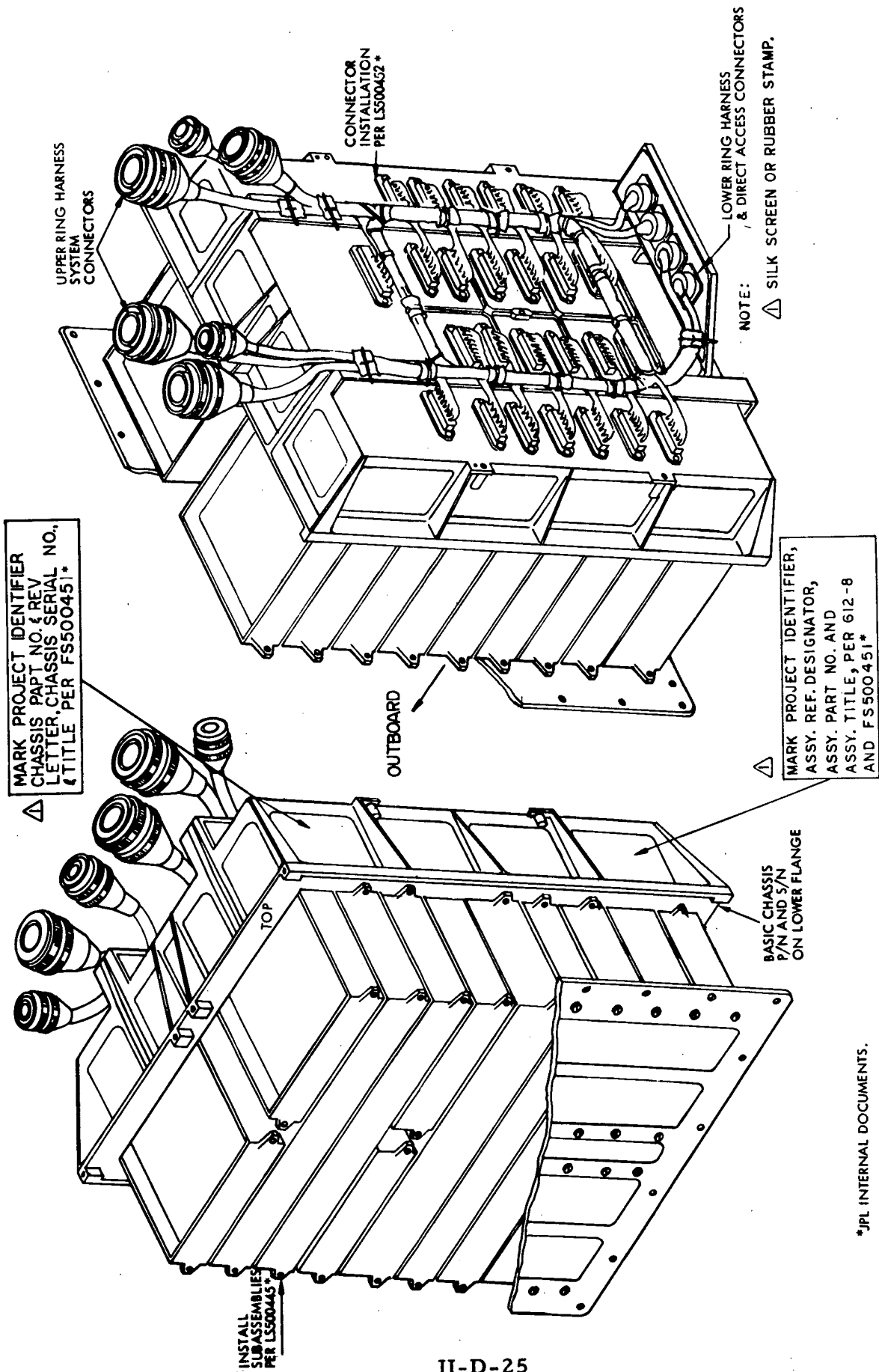
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\*JPL internal document.



\*JPL INTERNAL DOCUMENTS.

Fig. II-D-8. Typical Integral Shearplate Chassis Type Electronic Assembly



\*JPL INTERNAL DOCUMENTS.

Fig. II-D-9. Typical Plug-in Subchassis Type Electronic Assembly

electronic assemblies are contained within Section E. 7, Structure Subsystem.

- (b) The electronic assembly chassis shall have structural characteristics such that the resulting dynamic environment imposed upon the electronic subassemblies is compatible with all those approved electronic packaging techniques which are utilized. The outboard surface element of each assembly shall be designed to function as a primary shearplate when mounted on the primary structure; all holes, discontinuities and loads at attachment points in this outboard shearplate shall be controlled in order to assure structural integrity. The subchassis of each subassembly shall function as a load-bearing member and be mechanically integrated into the assembly structure.
- (c) The subassembly subchassis shall be designed as a mechanically integrated, load-bearing member of the electronic assembly. Adequate stiffness shall be provided to ensure that fragile parts, modules and interconnections are not damaged by normal levels of shock and vibration.

3) Thermal Requirements. Consistent with special equipment requirements, the electronic assemblies shall be designed to provide conductive heat paths to the structure. Subassemblies within each assembly will serve as thermal conductors. Subassemblies with high heat dissipation should be distributed to assist in the temperature control of all elements within the electronic assembly. Adjacent subassemblies shall make reciprocal use of radiative and conductive heat transfer to the maximum extent consistent with other factors. The basic assembly chassis shall provide a surface suitable for application of required temperature control finishes. Surface flatness and the



number of fasteners used shall be compatible with temperature control design requirements.

4) Interface Definition. The mechanical interfaces of the electronic assemblies shall be documented by interface control drawings, for all equipment that is designed by JPL contractors or experimenters. All cabling mechanical interfaces will be documented by cable installation drawings.

5) Performance Parameters. The design of the mechanical elements of electronic equipment shall ensure that a minimum of thermal and mechanical stress is imposed on electrical parts within the total range of SEP environments. In order to ensure adequate decoupling from the launch vehicle and the structure resonant frequencies during launch, the fundamental resonant frequency of electronic assemblies shall be no lower than 325 Hz in any axis. The fundamental resonant frequency of subassemblies in any axis shall be greater than 400 Hz in order to achieve additional mechanical decoupling and to control deflections of surfaces immediately related to component parts and their associated interconnections. The gains at resonant frequencies shall be less than 30, measured at discrete component parts relative to the primary structure/electronic assembly interfaces with an input vibration level of 5 g's.

6) Physical Characteristics and Constraints

(a) Weight and Volume

Electronic equipment shall be designed for minimum weight and volume consistent with high reliability, ease of fabrication, environmental stress, handling, durability, operational considerations, and flexibility to permit modification and/or rework without significant degradation.

E. SPACECRAFT SUBSYSTEM DESCRIPTIONS

1. Power Subsystem

The function of the spacecraft power subsystem is to receive power from the SEP module and to condition all power necessary for the spacecraft engineering and science subsystems. The spacecraft power subsystem is comprised of primarily VO'75 elements. Departures from VO'75 are made either to eliminate functional capability, existent in the VO'75 design which is not required; or to add capability required by the SEP module and mission requirements. A block diagram of the power subsystem is shown in Fig. II-E-1.

a. Power Sources

The primary power source is the SEP module solar arrays. Power is delivered to the spacecraft power subsystem via a pre-regulator in the SEP module. One NiCad 30 ampere-hour battery is utilized as a secondary power source in the SEP module to be used during periods when solar array power is not available.

b. Power Source and Logic

The power source and logic accepts power from all the power sources and controls the power distribution to the power conditioning units. To perform the control functions, a number of switches are employed.

1) Motor-Driven Switch. The motor-driven switch provides the switching from ground power during the prelaunch mode to spacecraft battery power just prior to launch. During ground operations, the switch provides dc power from the power support equipment (SE) to the spacecraft. The switching action is such that connection is made between the battery and the bus before it is broken between the bus and the external source (make before break). This ensures that the supply of power to the spacecraft is not interrupted during the switching process.

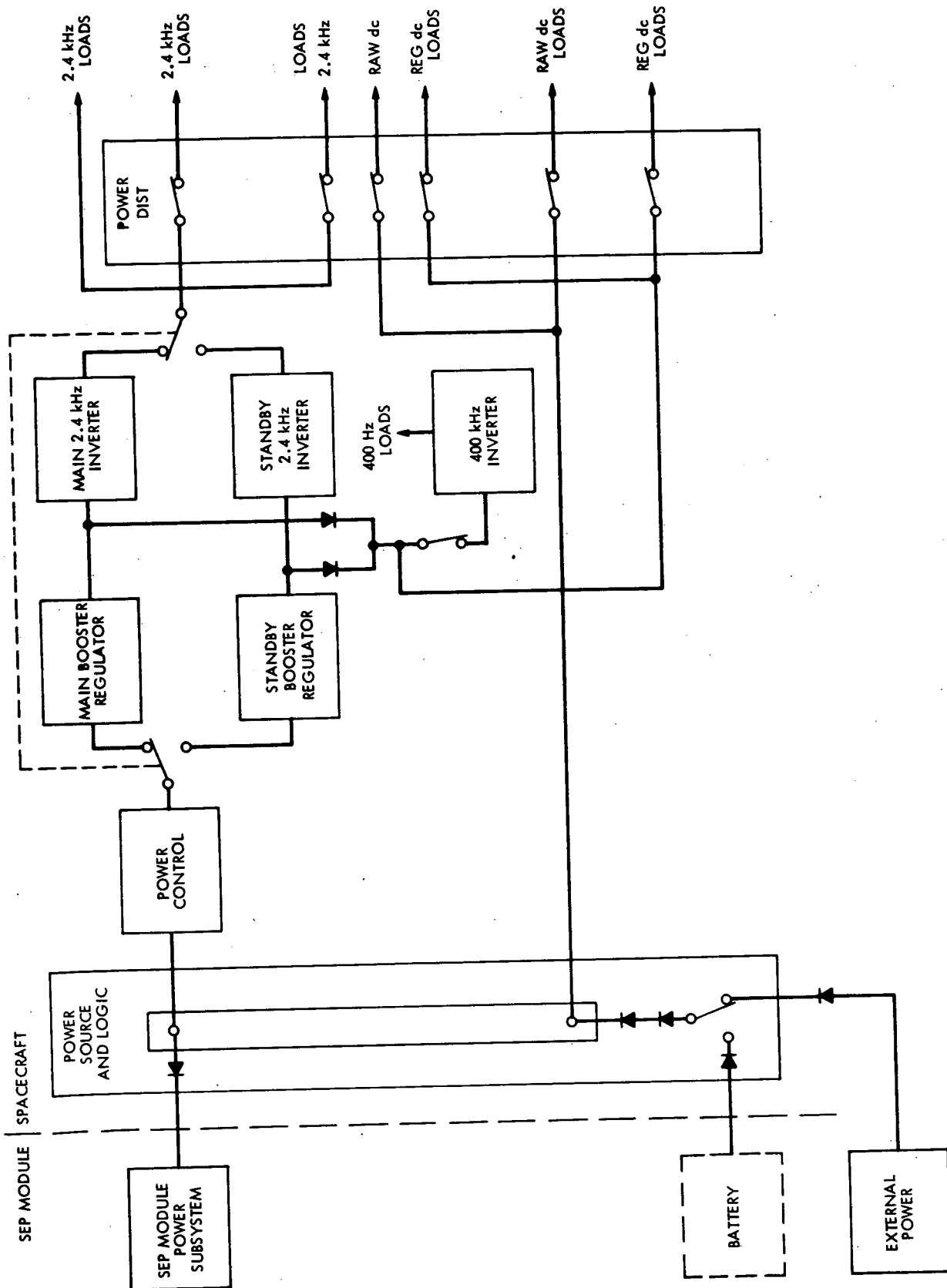


Fig. II-E-1. Viking Based Power Subsystem for SEP Spacecraft

2) Failure Sensor and Power Control Relay. The function of the failure sensor is to switch the main power bus and 2.4 kHz loads from the main booster regulator and main inverter to the standby booster regulator and standby inverter should a failure be detected. The output of the failure sensor controls the setting of the power control relay. The transfer to standby power disconnects the unregulated dc output of the power source logic from the input to the main power chain (main booster regulator and main 2.4 kHz inverter) and connects it to the input of the standby power chain, and it also connects the output of the standby power chain to the 2.4 kHz power bus and disconnects the output of the main power chain.

c. Power Conditioning

1) Main/Standby Booster Regulators. There are two booster regulators in the power subsystem. These boost the bus voltage to a regulated 56 V dc  $\pm 1$  percent. The main booster regulator handles all spacecraft 56 V dc power demands, and the standby booster provides a backup.

A booster regulator consists of four major parts: an input filter, an error amplifier, a transistor controlled autotransformer, and an output filter. Since the booster draws current in pulses, the input filter smooths high-frequency voltage variations which would occur in the raw power input. The error amplifier compares the output voltage with an internal zener voltage reference and generates an error signal. This signal acts through transistors to control a saturable autotransformer so that the autotransformer duty cycle increases with the error. An increased duty cycle results in a higher average regulated voltage. The output filter smooths the varying voltage at the autotransformer output so that the regulated voltage is a nearly constant 56 V dc.

2) Main/Standby 2.4 kHz Inverters. The main inverter supplies 50 V rms, 2.4 kHz power to the spacecraft subsystems. Input power comes from the 56 V dc output of the main booster regulator. The 2.4 kHz sync signal from the internal crystal and countdown chain provides a frequency

reference. In the absence of this sync signal, the main inverter will free run at 2.8 kHz, and this failure frequency will switch a relay to the standby power chain. The standby 2.4 kHz inverter is identical to the main inverter except that its free-running frequency is  $2.4 \text{ kHz} \pm 6 \text{ percent}$ . It receives input power from the standby booster regulator and sync pulses from its own internal crystal and countdown chain. The output of the standby inverter is routed through a contact of the relay to the same loads as supplied by the main inverter. The switching of the relay from main power to standby power is controlled by a failure sensor.

3) Three-Phase 400 Hz Inverter. The three-phase, 400-Hz inverter supplies 27.2 V rms three-phase, quasi-squarewave power to the attitude control gyro spin motors. Input power to the inverter is the redundant 56 V dc output of the booster regulators. The three-phase sync signals are provided by the 2.4 kHz power bus. Two identical inverters are used and are enabled or disabled by command.

4) 30 VDC Converter. The 30 V dc converter provides regulated dc to attitude control and propulsion functions. Two identical converters are used and are enabled or disabled by command.

d. Power Distribution and Switching

The power distribution and switching module consists of a number relays to switch power to the various distribution busses. There are four power distribution busses.

2. Attitude Control

The spacecraft attitude control system is essentially a Viking type high pressure nitrogen reaction control system and will be referred to as the RCS. The RCS is used during the ion engine-off phase of the mission. The RCS operates in the following modes:

- (a) Cruise mode
- (b) Acquisition mode
- (c) Inertial mode

The reaction control subsystem block diagram appears in Fig. II-E-2. The basic spacecraft components of the reaction control subsystem, shown in Fig. II-E-2 are:

- (a) Reaction control electronics
- (b) Star tracker
- (c) Sun sensors (bus mounted)
- (d) Reaction control assembly
- (e) Gyro package

In addition, the solar array mounted cruise sun sensors are used with the spacecraft reaction control system.

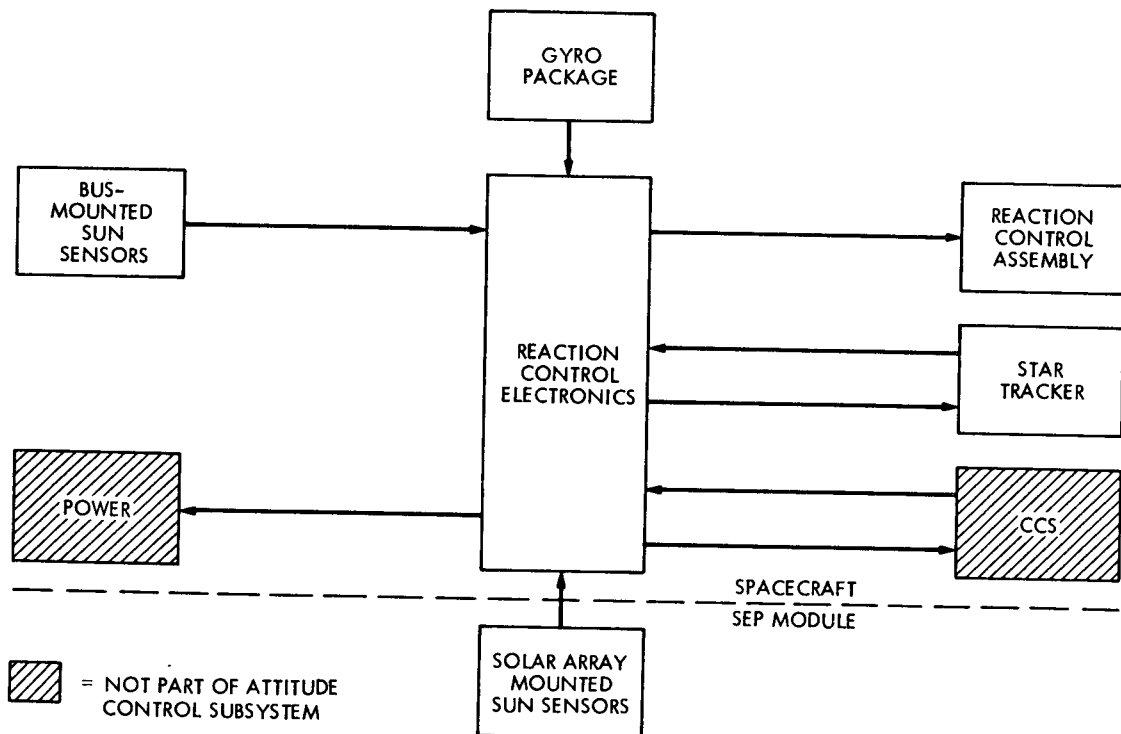


Fig. II-E-2. Reaction Control Subsystem Block Diagram

a. Basic Spacecraft Components Description

A brief description of the above components and their functions follows:

1) Reaction Control Electronics. The attitude control electronics assembly contains the electronic circuits required to control cold gas system torquing; all logic for the RCS, with the exception of that contained in the Canopus tracker; input buffering modules for most commanded inputs to the RCS; electrical interface with other spacecraft subsystems; power conditioning equipment; the rate estimator circuitry; and a number of other minor circuits.

2) Star Tracker. The star tracker is the position sensing element in the roll attitude control loop. It consists of an image dissector tube with a photocathode surface and associated optics and electronics.

The star tracker performs two primary functions. It makes an identification decision on each star that enters the field of view during roll search and provides a signal proportional to the roll error angle to the attitude control electronics when a star is identified as the reference star. The identification process consists of measuring the star intensity and comparing it with a previously calibrated value. Error angle information is obtained by repetitively scanning a slit field of view across the image field and then measuring the modulation phase of any star signal that appears.

3) Sun Sensors. The sun sensor assembly consists of a number of photoconductive cells and associated electrical circuits. The assembly provides the sensing function required for pitch and yaw attitude control of the spacecraft. Three types of sensors are used; acquisition sensors, cruise sensors, and sun gate.

When the spacecraft is operating in the acquisition mode the acquisition and cruise sensors operate together to bring the spacecraft into the desired orientation relative to the sun. The sun gate provides an

output which is converted into a logical signal to mark the completion of sun acquisition. The acquisition sensors are then disabled and the cruise sensors supply the information required to maintain the proper spacecraft-sun orientation throughout the cruise portion of the mission.

4) Gyro Package. The gyro package consists of three single-degree-of-freedom gyroscopes, and associated electronics. The purpose of the gyro package is to provide a means of sensing changes in spacecraft attitude when the celestial references are not acquired.

b. Spacecraft Control System Operation

In simplest terms, the spacecraft control system operates as follows: Attitude position and rate information sensed by the celestial sensor and rate estimator (during cruise) or rate gyro assembly (during commanded turns), is sent to the reaction control electronics and processed after which an appropriate signal is dispatched to the reaction control assembly where the appropriate control torque is exerted on the spacecraft by pulsed expulsion of gas.

Figure II-E-3 is a conceptual block diagram of the pitch, yaw and roll control in the acquisition and cruise modes. The block diagram for control of inertial and commanded turns is shown in Fig. II-E-4.

c. Sequence of Attitude Control Events

Current thought calls for the following sequence of attitude control events:

- (1) Separation
- (2) Attitude control system turned on

Sun acquisition. Pitch and yaw sensors and the gyro package are energized when RCS system is turned on.



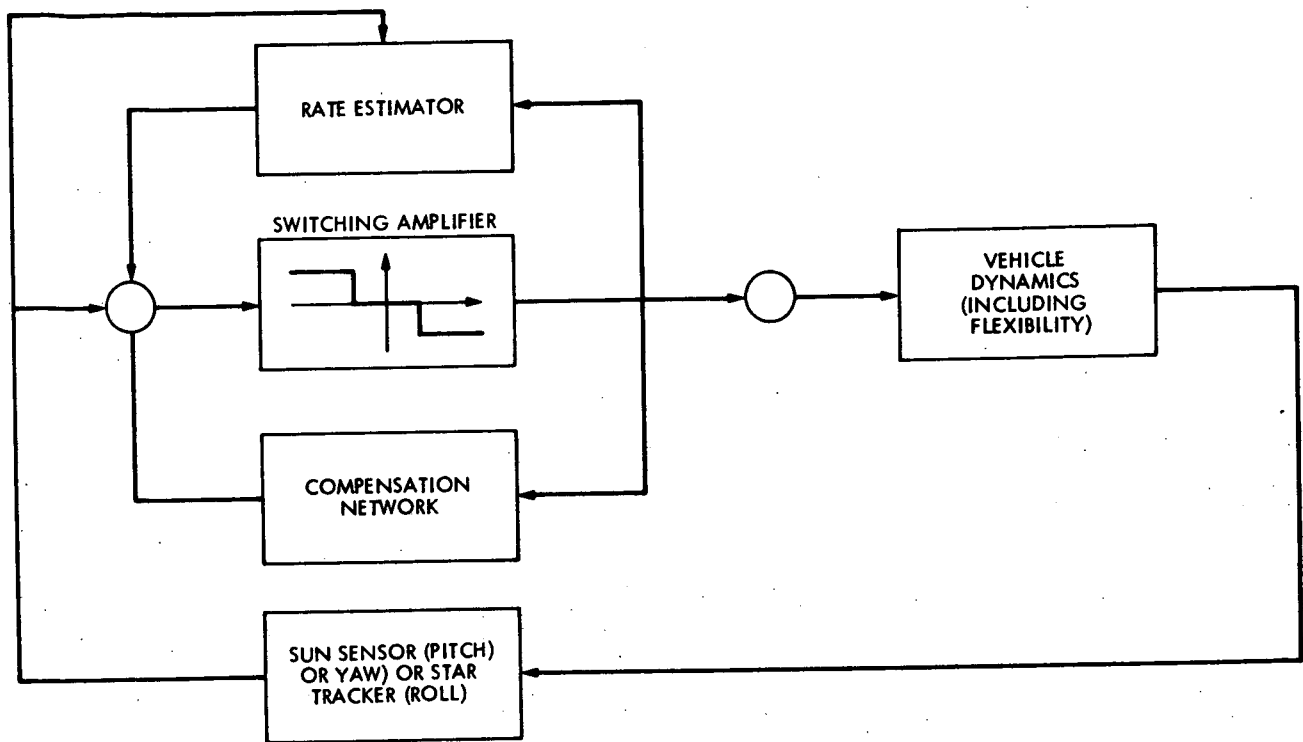


Fig. II-E-3. Pitch, Roll and Yaw Control Conceptual Block Diagram

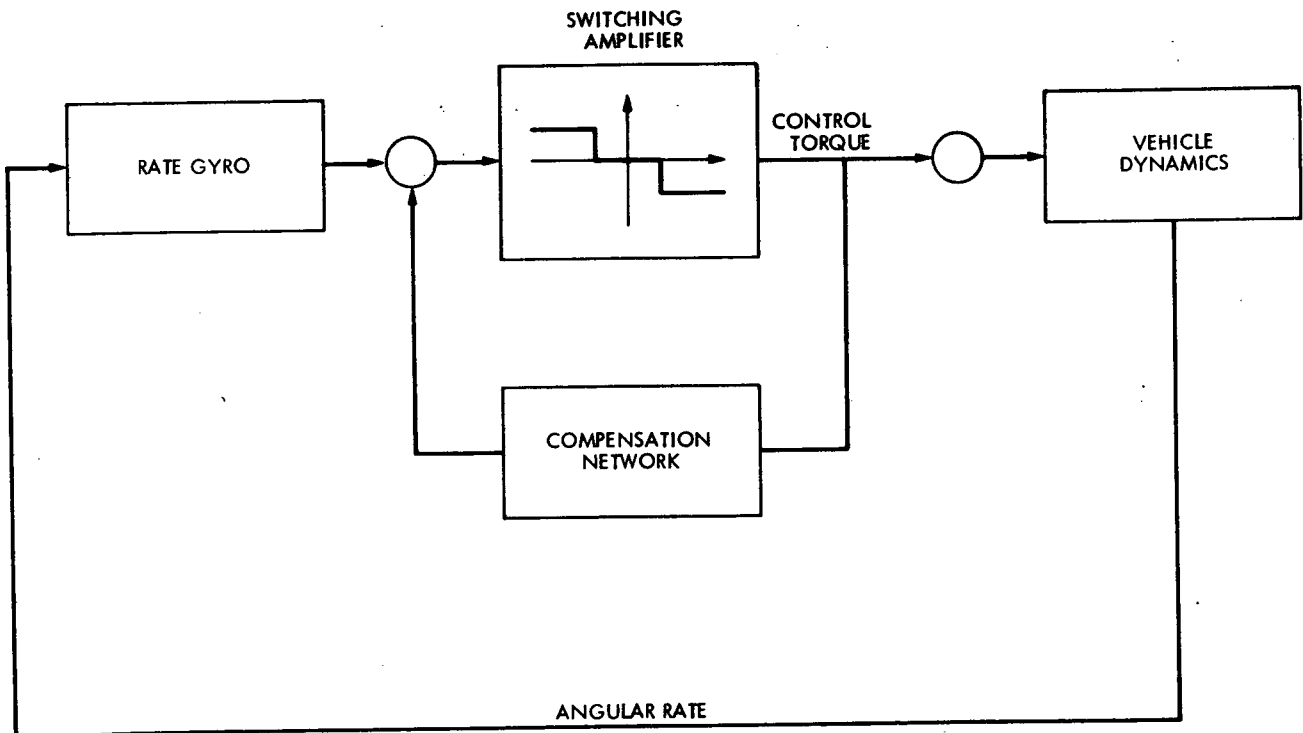


Fig. II-E-4. Conceptual Block Diagram for Control of Inertial and Commanded Turns

Sun is acquired by using position information from the sun sensors, rate information from the gyro package or rate estimator and control torques from the reaction control assembly.

(3) Reference Star Acquisition

- (a) Star tracker is energized by CCS signal. Initial turn-on after launch requires a suitable time delay to be safe from corona discharge.
- (b) Roll search initiated by tracker.
- (c) Reference star acquired using position information from the gyro package and control torques from the reaction control assembly. Because of field-of-view problems from the solar arrays, the star tracker must be capable of seeing reference stars other than Canopus. Electronic and mechanical biasing is provided for this purpose.

(4) Deployment of solar arrays

- (5) Desired attitude is maintained during limit cycle mode by using position information from the celestial sensors, passive damping from the derived rate networks, and control torques from the reaction control assembly.
- (6) Large angle turn of the vehicle performed prior to starting the electric propulsion engines.

d. Reacquisition After Disturbance

- (1) Turn off reference star tracker.
- (2) Gimbal sun sensors to nominal configuration.
- (3) Gimbal star tracker to orientation enabling it to track an appropriate reference star for this point in the trajectory.

- (4) Turn off thrusters and TVC (if in that mode), turn on RCS.
- (5) Proceed with normal sun and star acquisition.

3. Telecommunications Subsystem

The SEP spacecraft telecommunication subsystem provides for all communications required between the spacecraft and Earth after launch. The functions performed by the subsystem are as follows:

a. Carrier Tracking

The spacecraft receives a modulated or unmodulated S-band carrier from the ground station (DSIF). This received carrier is used to coherently excite the downlink at either S- or X-band frequencies (or both) such that the transmitted S- and X-band carrier frequencies shall be 240/221 and 880/221 times the received carrier frequency, respectively. When the spacecraft receiver has not acquired an uplink carrier, the downlink carrier is derived from a free-running crystal oscillator.

b. Ranging

The spacecraft provides the capability for S- and X-band planetary ranging. Uplink ranging data is demodulated from the S-band carrier and remodulated on either the S- or X-band downlink carriers (or both).

c. Commanding

The spacecraft provides for S-band command reception at 4 bps. Command data is received uncoded and biphase modulated on a square-wave subcarrier of 512 Hz. Command data reception is provided with a bit error rate of  $10^{-5}$ , or less.

d. Telemetry

The spacecraft provides the capability for transmission of telemetry data via S-band. Separate channels are provided for engineering and science data with block coding provided on the science channel. For data rates up to 10 kbps the subcarrier frequency is approximately 60 kHz. At data rates above 10 kbps, the subcarrier frequency is approximately 600 kHz.

e. S/X-Band Experiment

The spacecraft provides the capability for an S/X-band experiment consisting, as a minimum, of DSS downlink X-band Doppler tracking simultaneously with two-way S-band Doppler tracking, where S- and X-band downlinks are coherent. In addition, absolute range and/or DRVID (differenced range versus integrated Doppler) shall be possible over the S-band uplink/X-band downlink and/or the two-way S-band link, where link performance permits.

f. SEP Telecommunications Implementation (Refer to Fig. II-E-5)

The primary elements supporting the telecommunication functions are:

- (a) Radio frequency subsystem (RFS)
- (b) Modulation/demodulation subsystem (MDS)
- (c) S/X-band antenna subsystem (SXA)
- (d) X-band transmitter subsystem (XTX).

1) Radio Frequency Subsystem. The RFS shall have redundant receivers to enhance the reliability of the subsystem. An interface shall be provided supplying the XTX with a carrier reference signal and with the demodulated ranging signal. The receivers shall be of double-conversion superheterodyne design employing a phase-locked carrier tracking loop. The receiver shall perform the functions of command signal demodulation, ranging

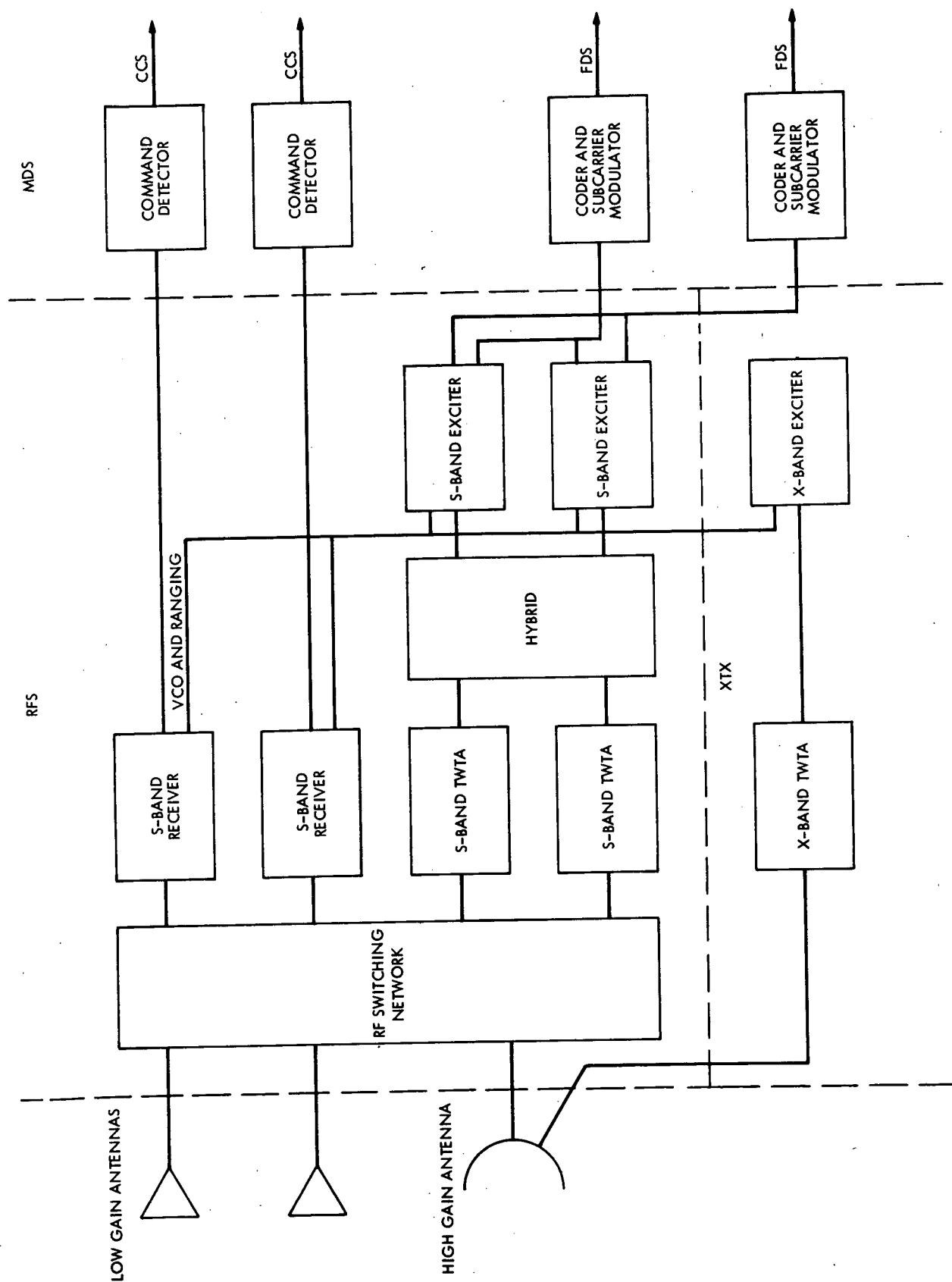


Fig. II-E-5. Telecommunications System Block Diagram

signal demodulation and carrier tracking. Selection between the dual redundant receivers shall be made by ground command or by programmed command. The programmed command shall initiate switching between receivers at a predetermined time. If a receiver switch is not desired, the programmed command shall be updated to occur at a later time. Each receiver shall be uniquely identified with one of the two redundant command detectors of the MDS providing the demodulated composite command signal to that detector. Power shall be supplied to the appropriate command detector of the MDS by the RFS.

The RFS shall provide dual redundant transmitters, each consisting of an S-band exciter and a traveling wave tube amplifier (TWTA). The redundant exciters shall be cross-coupled to the redundant TWTA's. Output power from either of the TWTA's shall be deliverable to either of the two antenna ports at the RFS/SXA interface by means of command selection. Selection of exciters and TWTA's shall be made by ground command or by CCS cyclic command. The CCS cyclic command shall initiate switching of exciters and/or TWTA's only when the respective subassemblies exhibit loss of power output at the time of the cyclic.

When the receiver has acquired the uplink carrier the phase reference for the transmitted carrier shall be derived from the carrier tracking loop. The phase of the transmitted carrier shall then be 240/221 times that of the received carrier. When the receiver has not acquired an uplink carrier the transmitter phase reference shall be derived from a free-running crystal oscillator.

The transmitter shall phase modulate the carrier with the telemetry signal received from the MDS/RFS interface. Additionally, the transmitter shall phase modulate the carrier with the demodulated ranging signal from the receiver whenever the ranging channel has been commanded "on" by ground command. Ranging modulation on the transmitted carrier shall cease upon reception of the CCS cyclic command provided for transmitter failure switching. Reinitiation of ranging on the transmitted carrier shall then be possible only by ground command.

2) Modulation/Demodulation Subsystem. The MDS shall perform the functions of data modulation and detection within the spacecraft. The MDS shall be implemented so as to

- (a) Receive telemetry data from the FDS, code the data for efficient channel transmission, biphasic modulate the coded data on a squarewave subcarrier.
- (b) The MDS shall receive a composite command signal from the RFS/MDS interface consisting of a squarewave subcarrier at 512-Hz biphasic modulated with command data and bit synchronization plus noise; acquire synchronization with the subcarrier and data signals, estimate data bit values within the data stream and deliver the data to the MDS/CCS interface.

The data modulation and data detection functions of the MDS shall be independent and each shall be implemented with dual redundant equipment. Each of the redundant telemetry modulators shall be selectable by ground command issued through CCS to the power subsystem. Each of the redundant command detectors shall be uniquely associated with one of the redundant receivers within the RFS and shall receive its operating power from the RFS. A particular command detector shall be powered only when its associated receiver in the RFS is operating.

3) S/X-Band Antenna Subsystem. The SXA shall be configured with one high-gain antenna (HGA) and two low-gain antennas (LGA), one forward and one aft.

The HGA shall be a circularly symmetrical parabolic reflector 1.47 meters in diameter with a focal point feed assembly for S- and X-band transmission. The boresight of the HGA shall be adjustable in flight with two degrees of freedom. The HGA is used for S-band transmission and

reception, and X-band transmission. The LGA's shall provide for S-band transmission and reception and shall provide a broad-beam roll symmetric pattern with boresights along the spacecraft Z axis (in the direction of the sun). Such an orientation shall provide the basic receiving pattern necessary for earth-spacecraft communications in sun-acquired attitudes.

4) X-Band Transmitter Subsystem. The XTX shall transmit a phase-modulated, X-band carrier and deliver it to the XTX-SXA interface. The implementation of the XTX shall be such that it produces a carrier frequency of exactly 440 times the frequency received from the RFS/XTX interface. This shall result in a transmitted carrier that is phase coherent with the S-band transmitted carrier of the RFS. When the RFS has acquired two-way lock to an uplink carrier the resultant X-band carrier frequency shall be 880/221 times the uplink carrier frequency. The XTX shall modulate the carrier with ranging information received from the RFS/SXA interface, when present.

XTX on/off control shall be exercised by the power subsystem and initiated by ground command. The ranging signal shall be supplied to the XTX from the RFS whenever the ranging channel has been commanded on within the RFS.

5) Performance. The performance of the various functions of the telecommunications system as outlined here are described in Figs. II-E-6, II-E-7 and II-E-8. It is expected that the communications blackouts will occur around 320 and 720 days from launch. At these times, the sun-earth-space vehicle angle becomes less than 5 degrees, severely degrading link performance.

#### 4. Computer Command Subsystem

The function of the computer command subsystem (CCS) is to issue commands to the spacecraft and SEP module, and to generate commands for the SEP module based on SEP module status data.



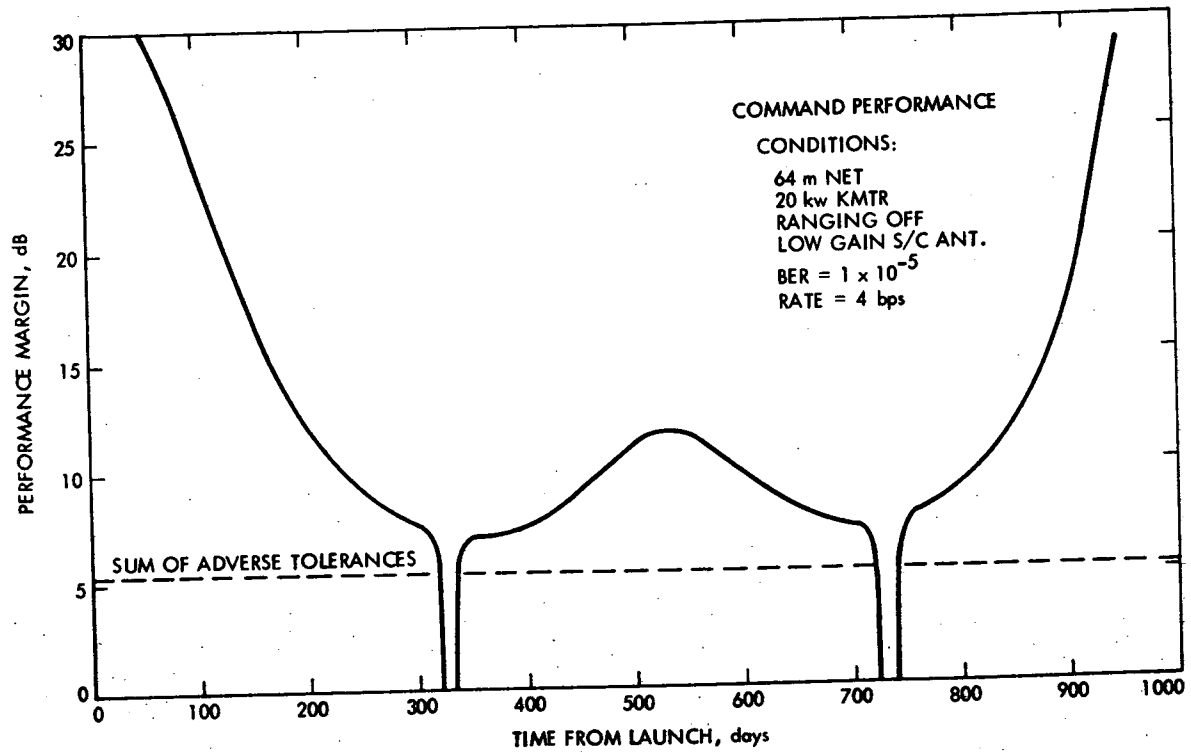


Fig. II-E-6. Command Performance Margin

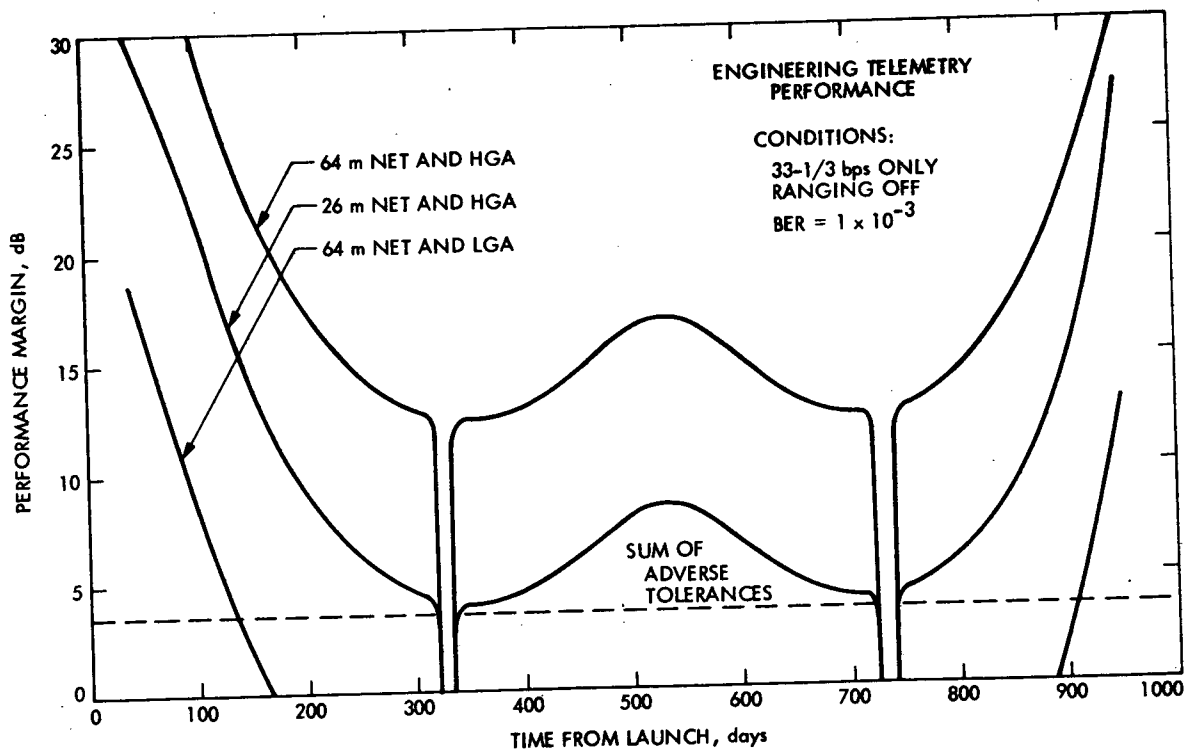


Fig. II-E-7. Telemetry Performance Margin

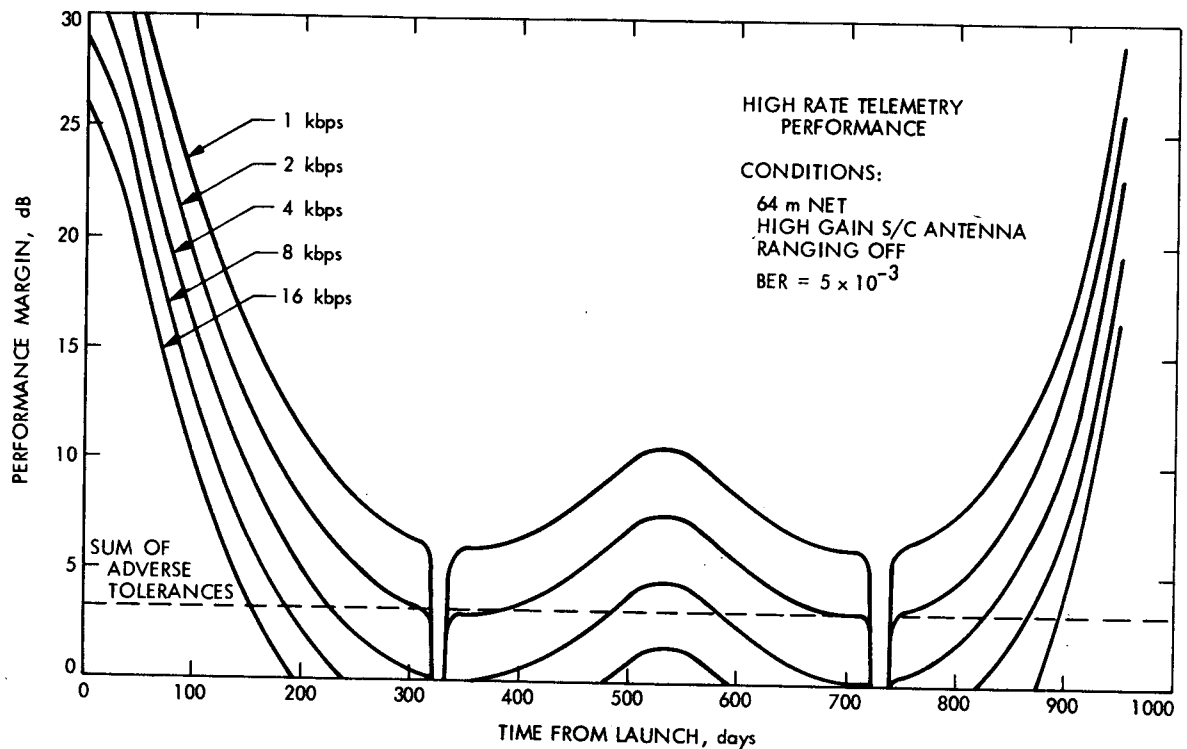


Fig. II-E-8. High Rate Telemetry Performance

a. Description

The CCS is identical to the VO'75 CCS. It consists of six functional units as illustrated in the hardware functional block diagram, Fig. II-E-9. Although the inputs and outputs are shown in pairs, many of the pairs are "wire or's" in the CCS harness to reduce spacecraft wiring.

b. Processors

The processor is the most complex of the CCS functional units and can be functionally subdivided as illustrated in the processor functional block diagram, Fig. II-E-10.

The central processor either processes instructions or waits for an interrupt. An interrupt causes an instruction to be accessed from a

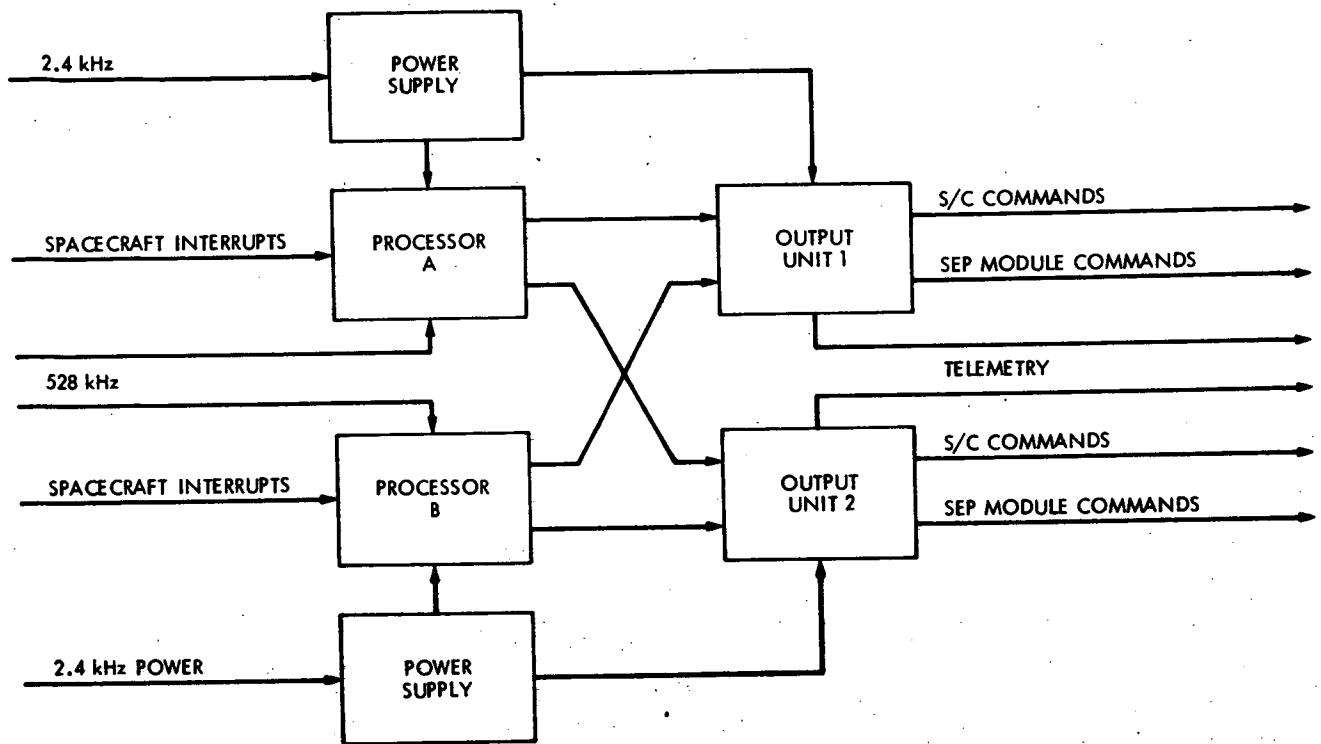


Fig. II-E-9. CCS Hardware Functional Block Diagram

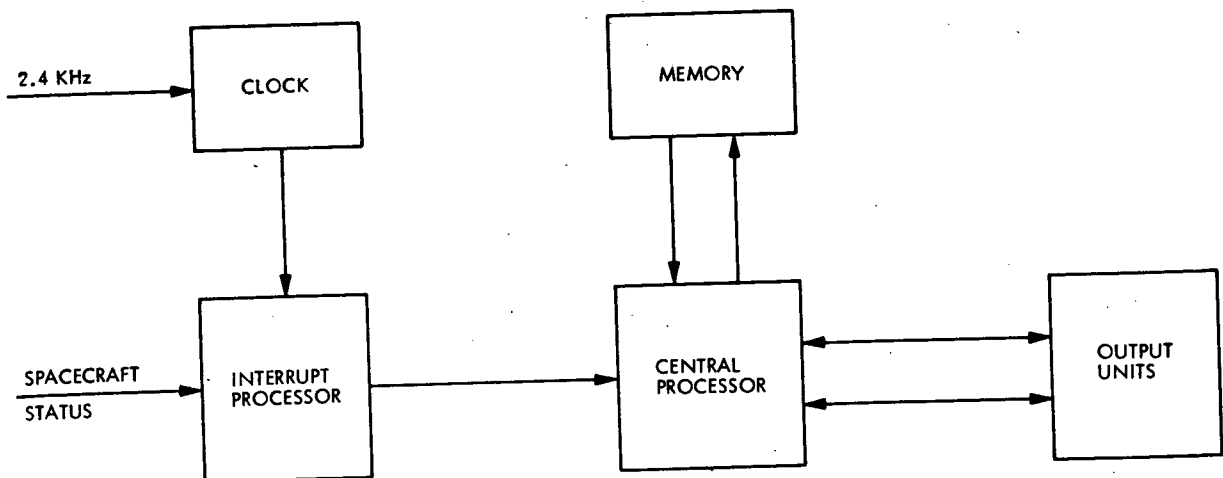


Fig. II-E-10. CCS Processor Functional Block Diagram

memory location unique to that interrupt. New interrupts can be responded to immediately, or after the current processing has been completed.

The interrupt processor, using a maskable priority structure, informs the central processor as to which interrupt shall be processed next. The interrupt processor is alerted to spacecraft, CCS subsystem, and ground command status through interrupts. The interrupt processor also relays status information to the central processor upon request.

The processors operate in the following modes:

- (1) Tandem: Both processors are active simultaneously, and work in series.
- (2) Parallel: Both processors are simultaneously performing the same functions.
- (3) Separate: Both processors are active and performing different functions.

The memory stores instructions, and data for those instructions. It transfers this information to and from the central processor upon command. The first half of the 4096-word capacity is under write-protect control to allow it to be effectively used as a read-only memory.

The clock is synchronized to the flight data subsystem through the power subsystem. From this clock the interrupt processor receives the following three interrupts:

- (1) One pulse every hour.
- (2) One pulse every second.
- (3) One pulse every ten milliseconds.

c. Output Units

The output units have three basic functions: issuing output commands, providing telemetry data to the FDS, and providing communication with the SEP module master FDS.

The output units convert output command data which they receive from either of the two processors to output commands which are recognizable by the user subsystems. These output commands take one of two forms:

- (1) Pulsed discrete commands
- (2) Short duration closing of isolated interface switches

The output units provide telemetry data to the FDS. Functionally, the CCS has six output registers for telemetry, three registers associated with each output unit. Each register has a 28-bit storage. Seven bits of the register in use is shifted out to FDS at each commutation. The output registers may contain one of three types of words:

- (1) Processor Word: Highest priority
- (2) Output Event Word: Middle priority
- (3) CCS Status Word: Lowest priority

The Buffer units provide three necessary functions as follows:

- (1) Decoding for the individual interfaces.
- (2) D. C. isolation between the CCS and the spacecraft users.
- (3) Low level to high level signal conversion.

SEP module communications is performed at the processor clock rate. The output unit provides single-bit buffering and dc isolation.

d. Power Supplies

Besides providing the voltages necessary to operate the CCS functional units, the power supplies inhibit the functional units from operation during periods when the power supplied is not adequate, inform the processors when power supplied is not adequate, and insure that the power bus will not be shorted out due to a failure internal to the CCS.

e. Interfaces

The CCS receives primary 2.4 kHz square-wave power. The MDS supplies the CCS with a command channel from each of its command detector units (CDU). Each channel consists of command data, bit timing pulses, and command initializing pulses.

The FDS supplies the CCS with the inputs necessary for two digital telemetry channels. Correspondingly, the CCS shall supply required outputs. Each channel consists of a telemetry alert, bit sync input, and a telemetry data output.

From ACS, the CCS will receive two level inputs. One will indicate when the sun is acquired and the other will indicate when Canopus is acquired.

Subsystem users receiving digital data from the CCS shall do so over three types of interfaces as described below:

- (1) Data: A series of levels representing ones and zeros for fourteen bit times.
- (2) Strobe: Follows data immediately and indicates that all the data has been sent.
- (3) Enable: Brackets data and strobe to indicate an activity period.

The first of the fourteen data bits is a zero, except for attitude control subsystem's scan platform and positioning commands. The second of the fourteen data bits represents odd parity over all fourteen data bits. The remaining twelve data bits are user dependent.

The master FDS provides a data return line to the CCS. This line is used to inform CCS of problems that have been uncovered by the master FDS and as an information return line when the CCS requests data.

f. Performance Parameters

1) Timing Accuracy. The stability of the basic power subsystem reference from which CCS timing is derived is TBD. It is anticipated that accumulation of errors resulting from this basic frequency offset will be compensated for to a large extent under CCS program control, based on calibrated performance of the basic frequency source located in the FDS against the expected flight environment.

2) Bit Timing Accuracy. The stability of the basic FDS reference from which CCS bit timing is derived is TBD. Any accumulation of errors resulting from this basic frequency offset will be eliminated when the CCS enters an inactive state.

3) Resolution. The CCS has three timing inputs. From these, commands can be issued with the following resolutions:

- (1) One pulse per hour gives  $\pm 30$  minutes resolution.
- (2) One pulse per second gives  $\pm 500$  milliseconds resolution.
- (3) One hundred pulses per second gives  $\pm 5$  milliseconds resolution.

4) Ground Command Bit Rates. The command information bit rate received from the MDS is 4 bps.

5) CCS Command Rates. Commands can be issued consecutively by the CCS. It takes a maximum of 20 milliseconds to issue a coded command. It takes a maximum of 110 milliseconds to issue a discrete command. It takes a maximum of 200 microseconds to command the master FDS.

g. Physical Characteristics

1) Weight. The weight of the CCS exclusive of case harness and case shall not exceed 42 pounds.

2) Power. The average 2.4 kHz prime power required by the CCS shall not exceed 15 watts.

5. Flight Data Subsystem

The FDS has two primary functions. First, it provides the spacecraft with the primary source for central timing. Second, it controls, collects, and formats both science and engineering data. Reference frequencies are provided to the modulation/demodulation subsystem (MDS), data storage subsystem (DSS), power subsystem (PSS), and to those science instrument that require them.

The FDS controls and sequences the science instruments. Science and engineering data are collected and formatted into serial bit streams. The FDS performs analog-to-digital conversion, signal conditioning, and digital data processing, as required. Formatted data is sent to the MDS for real-time transmission to earth or to the DSS for temporary storage.

The FDS is divided into the following six major functional areas:

- (1) Timing and control block
- (2) Engineering data block
- (3) Non-imaging science data block
- (4) Visual image data block
- (5) Memory block
- (6) Power conversion block.

a. Timing and Control Block

The timing and control block includes the redundant oscillators, timing chains, and coded command decoding logic; the engineering sequencing and control logic; the science sequencing and control logic; and the power-on reset circuitry.



b. Engineering Data Block

The engineering data block handles both digital and analog measurements that can be made at relatively low rates and with no more than 7-bit accuracy. Reference voltages and currents for pressure and temperature measurements are generated by this block. Also contained in this area are the analog commutators (tree switches) that multiplex the data and references for the measurement and conditioning circuitry.

c. Non-Imaging Science

The non-imaging science (NIS) data block multiplexers assemble the NIS format. Two identical, redundant outputs are generated by separate FDS hardware to be transmitted to the DSS and also interlaced with the visual imaging data.

d. Visual Imaging Data Block

The visual imaging data block processes high-rate (2 MHz) data from the visual imaging subsystem (VIS) into 7 data tracks for the tape recorders in the DSS.

e. Memory Block

The memory block includes two identical plated wire memories. Each memory provides random access storage and retrieval of 1024 words of 8 bits each.

f. Power Conversion Block

The FDS includes redundant power converters for converting the 2.4 kHz input power to the specific filtered and regulated voltages required to operate the circuits, logic, relays and memories. Power output switching is also included in this block.

The basic approach to reliability for the FDS is the use of redundancy. Any block of logic that controls or is used in the transfer of data from more than one science instrument (including engineering) will be redundant. There will be no redundancy where logic is dedicated to a given science instrument.

6. Data Storage Subsystem

a. Design

The DSS consists of the Viking Orbiter '75 transport with electronics design based on past Mariner and Viking designs.

b. Description

The DSS is divided into two parts: the data storage subsystem electronics (DSSE) and the data storage subsystem transport (DSST). The DSSE includes most of the electronics functions of the DSS. The DSST consists of the tape-drive mechanism which includes heads, motor, end-of-tape (EOT) sensors, various environmental transducers and playback amplifiers.

The DSST is a reel-to-reel co-planar peripheral drive transport containing 1,000 feet of 1/2 inch, 1 mil magnetic tape. The head stack consists of nine in-line tracks, eight of which are used to store data; the ninth contains a pre-recorded signal used as the tachometer feedback for speed control. A hysteresis synchronous motor coupled through two pulleys for speed reduction serve as the prime mover of the transports. A block diagram of the DSS is shown in Fig. II-E-11, and the functional characteristics are summarized in Table II-E-1.

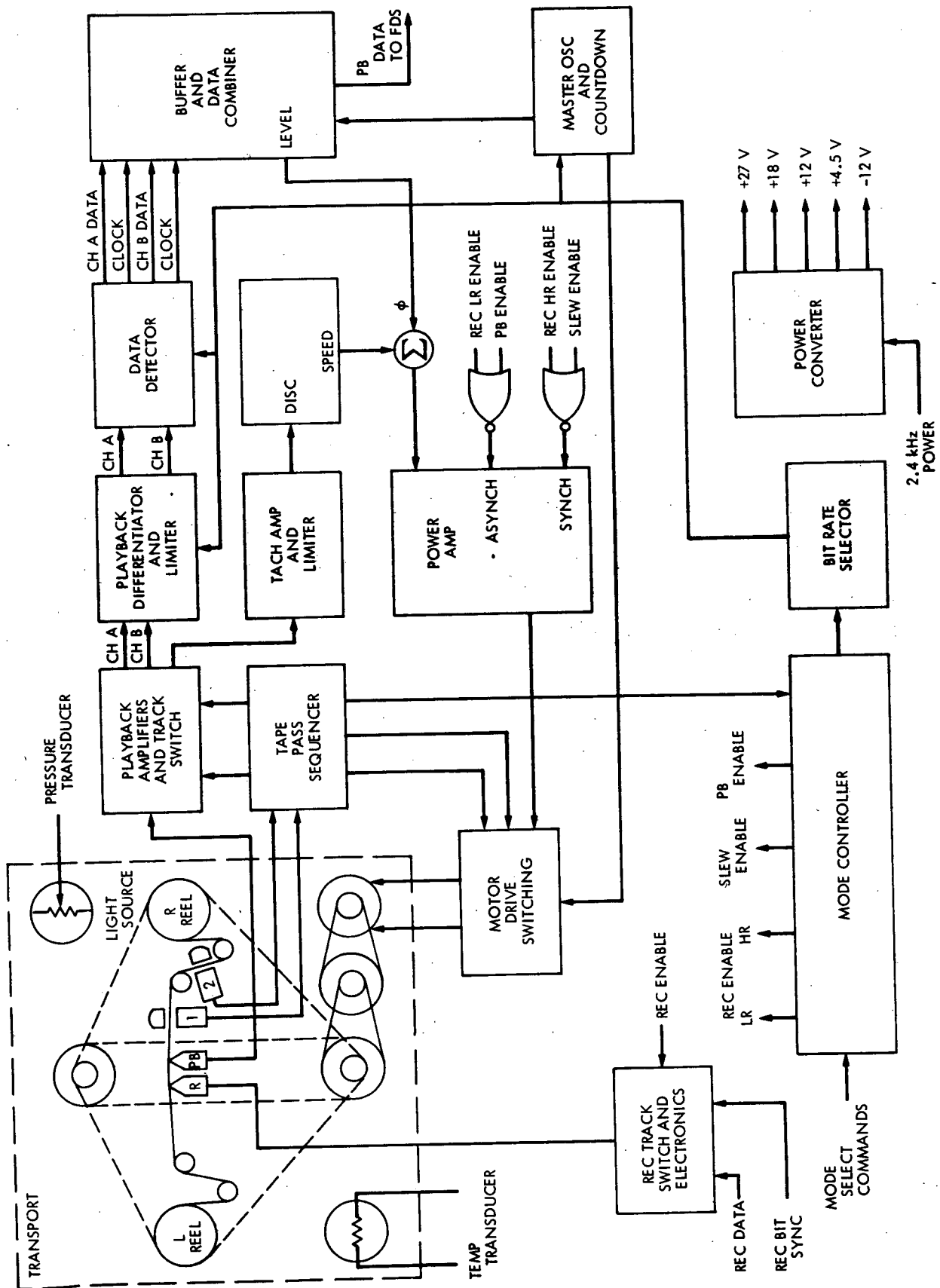


Fig. II-E-11. DSS Block Diagram

Table II-E-1. DSS Functional Characteristics

Parameter	Characteristic												
Transport configuration	Co-planar, reel-to-reel, peripheral drive												
Heads	9 tracks, hard tip (long life) in-line. Record and reproduce head stack.												
Tape size	1000 ft. of 1/2" tape												
Capacity	$2.3 \times 10^8$ bits												
Modes	Record high; record low; playback 1, 2, 3, 4, and 5; slew; ready.												
Record rates (tape speed)	<table> <tr> <th><u>kbps</u></th><th><u>ips</u></th></tr> <tr> <td>130</td><td>(27.1 )</td></tr> <tr> <td>1</td><td>( 0.21)</td></tr> </table>	<u>kbps</u>	<u>ips</u>	130	(27.1 )	1	( 0.21)						
<u>kbps</u>	<u>ips</u>												
130	(27.1 )												
1	( 0.21)												
Playback rates (tape speed)	<table> <tr> <th><u>kbps</u></th><th><u>ips</u></th></tr> <tr> <td>16</td><td>( 3.3 )</td></tr> <tr> <td>8</td><td>( 1.7 )</td></tr> <tr> <td>4</td><td>( 0.83)</td></tr> <tr> <td>2</td><td>( 0.42)</td></tr> <tr> <td>1</td><td>( 0.21)</td></tr> </table>	<u>kbps</u>	<u>ips</u>	16	( 3.3 )	8	( 1.7 )	4	( 0.83)	2	( 0.42)	1	( 0.21)
<u>kbps</u>	<u>ips</u>												
16	( 3.3 )												
8	( 1.7 )												
4	( 0.83)												
2	( 0.42)												
1	( 0.21)												
Data tracks	8												
Tachometer reference track	1												
Packing density	2400 bpi/track												
Track usage	Record or playback 2 tracks simultaneously												
Error rate	1 error in $10^3$ bits												
Load/unload cycles	350 max												
Power	<table> <tr> <td>Record</td><td>&lt; 30 watts</td></tr> <tr> <td>Slew</td><td>&lt; 29.5 watts</td></tr> <tr> <td>Playback</td><td>&lt; 16.5 watts</td></tr> <tr> <td>Ready</td><td>&lt; 12.5 watts</td></tr> </table>	Record	< 30 watts	Slew	< 29.5 watts	Playback	< 16.5 watts	Ready	< 12.5 watts				
Record	< 30 watts												
Slew	< 29.5 watts												
Playback	< 16.5 watts												
Ready	< 12.5 watts												
Weight	30.4 lb. excluding case harness and bay chassis												
Volume	1 full bay												

c. Modes of Operation

The DSS has four basic modes of operation. These are:

- (1) Record
- (2) Playback
- (3) Slew
- (4) Ready.

1) Record Mode. In the record mode, the DSS accepts one of two serial data streams supplied by the Flight Data System (FDS). The high data rate is 130 kbps video data and the low data rate is 1 kbps science data. These data are recorded on the tape at 27.1 ips and 0.21 ips, respectively.

The serial input data from the FDS is split and encoded into two data streams and recorded on two data tracks. The packing density is 2400 bpi. Record track sequencing is performed automatically by the DSS. Four tape passes are required to complete a record cycle, providing a storage capacity of  $2.3 \times 10^8$  bits.

The record sequence can be initiated by the FDS or by the CCS. Recording is automatically terminated when the DSS reaches EOT on the fourth pass, or upon receipt of a command from either the FDS or CCS.

2) Playback Mode. The DSS will play back the recorded data into the FDS at one of five pre-assigned bit rates; (Table II-E-1). Playback track sequencing is performed while playing back two tracks of data at a time, requiring four passes for a complete reproduce cycle.

The signals at the playback head stack are amplified by the playback preamplifiers, with sufficient gain such that track sequencing may be accomplished without significant degradation of signal-to-noise ratio. The two selected data tracks are further amplified, signal conditioned, and detected

by the data detectors. The extracted data and clocks are recombined into a serial data stream, and loaded into a circulating de-jitter buffer. The output of the buffer is clocked by a reference clock providing jitter-free data to the FDS.

The playback mode is initiated by either a CCS or FDS command, and is terminated by either a CCS or FDS command, or by an EOT signal at the completion of the playback cycle.

3) Slew Mode. The DSS slew mode is provided such that the tape may be positioned at synchronous record speed by ground commands without erasing the stored data. This mode may be used to position the tape prior to starting a record sequence, or to play back selected portions of stored data.

4) Ready Mode. In the ready mode, 2.4 kHz power is supplied to the DSS but the tape is not in motion and the record heads are not energized.

d. Speed Control

Tape speed control is accomplished during the high speed (high data rate) record or slew mode by operating the motor synchronously at approximately 7440 rpm. All other speeds are controlled in the asynchronous mode by regulating current to the motor. Two feedback loops are employed. The first loop derives a rate feedback signal from the output of the tachometer track playback amplifier. The tachometer track is pre-recorded on tape to provide a signal independent of the data tracks. This loop seeks to maintain tape velocity even during the absence of the recorded data. The second loop employs the data buffer level as it references inputs and acts to keep the buffer half full. The servo is mechanized such that during playback (except during tape direction reversal) the buffer capacity is not exceeded.

e. Mode Controller/Rate Selector

The mode controller generates logic signals in response to the FDS and CCS commands to enable the record, playback, slew and ready modes. The controller also provides applicable selected clock rates and signals to implement the desired record or playback rates. The rate selector provides countdown ratios for the various bit rates commanded.

7. Structure Subsystem (STRU)

The spacecraft primary structure consists of a skin-stringer box structure which serves as the spacecraft electronic compartment, and an open truss which supports the electronic compartment from the SEP module. The electronic compartment is designed with four Viking-type electronic bays on each of the two sides which are flown parallel to the ecliptic plane.

As shown in Fig. II-E-12, the structure of the electronic compartment provides support for the electronic chassis and consists of 6 longerons which connect upper, center and lower frames. The outer surfaces of the electronic chassis serve as shear plates in the structure and also provide meteoroid protection when used in conjunction with the louver assemblies. The remaining surfaces of the electronic compartment are also stiffened by shear panels and shear panel intercostals. These surfaces are also required to serve as meteoroid shields.

Besides supporting the spacecraft electronics, the electronic compartment serves as the spacecraft primary structural reference for communication, attitude control and science instrument alignment. In this role, it provides structural interfaces for the science platform, the antennas, and the star tracker. Attitude control propellant tanks are also supported from the electronic compartment.

In order to minimize weight, an open tubular truss assembly is utilized to support the spacecraft electronic compartment from the SEP module. The upper end of the truss is fastened to the electronic compartment during all

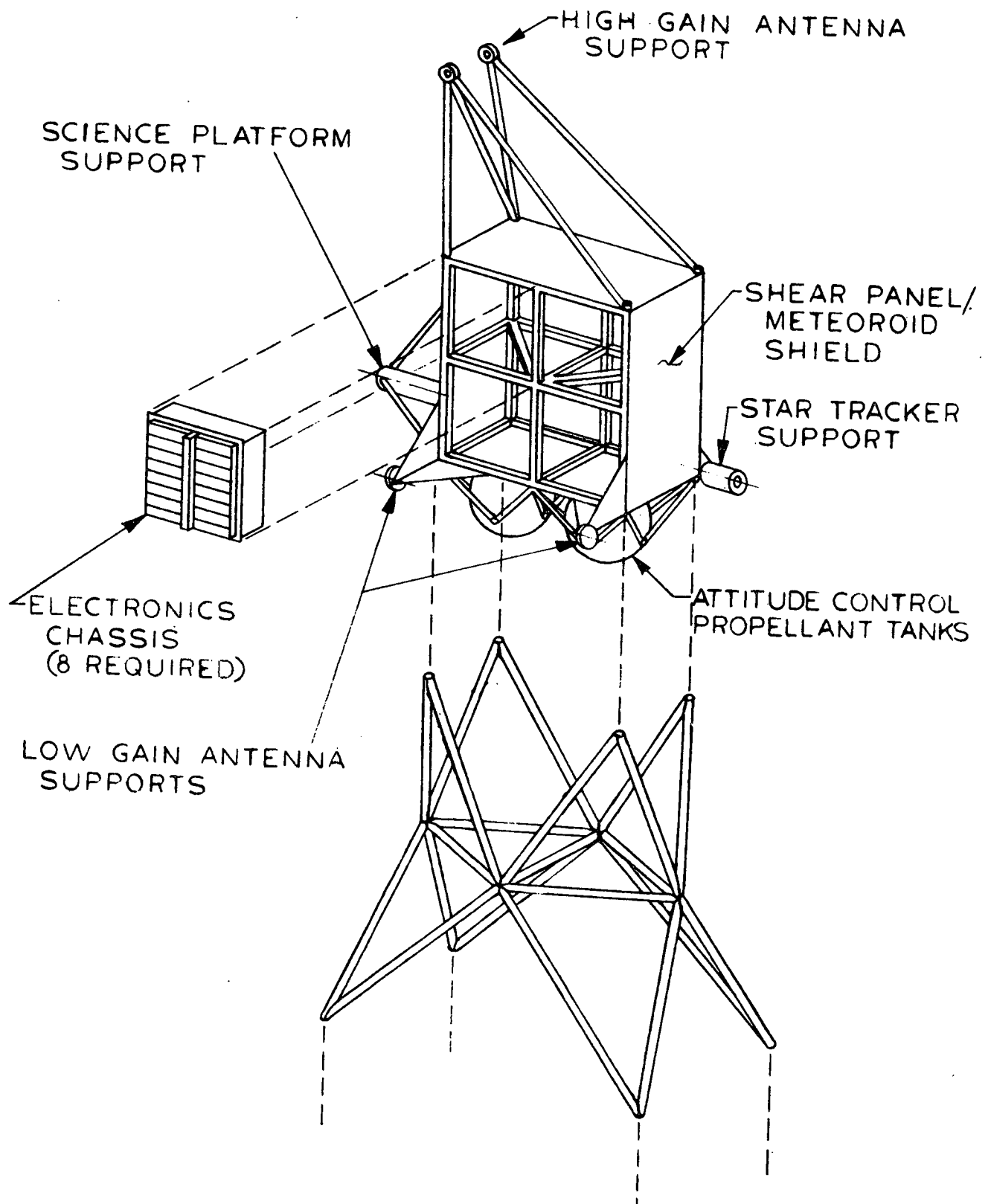


Fig. II-E-12. Spacecraft Structure



phases of the mission. The lower end of the truss is attached to the corner longerons of the SEP module with separable fasteners and is capable of being separated from the SEP module.

8. Cabling Subsystem (CABL)

a. Scope

The cabling subsystem provides electrical interconnections for all interfaces between flight subassemblies, assemblies, and subsystems, and for required interfaces from the spacecraft to the SEP module, the launch vehicle and launch support equipment. All cabling required to make these electrical connections on the flight spacecraft and its launch vehicle adapter is included in the cabling subsystem. Not included is wiring that is part of subassemblies, and rf coaxial cabling.

b. Design

The cabling design assumes maximum use of VO'75 design and technology. Major design features include the use of No. 26 AWG copper alloy conductor with TFE insulation and rectangular D Series or circular Bendix "JT" type connectors.

The flight electrical cabling is designed to:

- (1) Transmit the signals and power in a manner compatible with the requirements of the source unit and the destination unit.
- (2) Control induced electrical interference.
- (3) Control power loss.
- (4) Control voltage drop.
- (5) Shield wires carrying otherwise incompatible signals or power.
- (6) Provide adequate electrical insulation.

- (7) Provide convenient electrical connect/disconnect capability for installation and removal of each subsystem.

c. Description

The cabling subsystem will include approximately 20 cables. These cables are of three major types, as described below:

- 1) Electronic Assembly Harnesses. Eight harnesses interconnect the subassemblies of each subsystem, and provide connections for system level functions to the upper and lower RINGs. Connectors for "direct access" of test functions are provided on the lower bracket of the harness assembly, if required by the subsystem.

- 2) System RING Harnesses. The upper RING (remote interconnecting grouping) harness is contained within a support structure located within the primary structure. To control cross coupling, the design of the upper RING harness provides for physical separation of wire groups and, wherever possible, the associated connectors for power, noisy-signal and quiet-signal circuits. The connectors of the upper RING harness are either hard-mounted to the supporting structure or connected to hard-mounted connectors on the electronic assemblies. The connectors are accessible from the top of the spacecraft (with thermal shield removed).

If required, a lower RING harness shall be provided on the structure adjacent to the lower periphery of the electronic assemblies. Cable branches out of the lower RING harness interconnect the electronic assemblies with the spacecraft adapter and other electronic equipment located in the lower portion of the spacecraft.

- 3) External cables. Electrical cables are supplied as necessary for equipment not contained within the bays of the primary structure. Such cabling includes pyrotechnic distribution lines, attitude control gas-jet

wiring, temperature transducer wiring, cabling of the scan platform, cruise science, and intersystem cabling to the adapter and launch support equipment. The physical and electrical characteristics of special harnesses accommodate any special requirements of the equipment to be interconnected.

9. Mechanical Devices Subsystem (DEV)

The spacecraft mechanical devices subsystem consists of the following mechanisms:

a. High Gain Antenna (HGA) Latches and Deploy Mechanisms

The HGA latches shall restrain the HGA to the spacecraft structure in the launch position. The antenna dish will be restrained against the HGA boom to prevent launch loads from feeding back into the antenna articulation actuators. The HGA boom will be restrained to the spacecraft structure.

The HGA, upon command, shall deploy in a single continuous motion to a single deployed position. The deployment shall be provided by a spring-damper mechanical linked device.

At full deployment, the mechanism shall latch to properly position the HGA. The device shall provide an electrical switch closure to indicate full deployment.

b. Science Platform and Latch Mechanisms

The science platform shall be a single-support platform that shall be restrained to the spacecraft structure during launch. The latch shall be capable of release, on command, after spacecraft separation from the adapter.

c. Thermal Louvers and Control Mechanisms

The louvers shall be bolt-on assemblies with removable louver blades. The louver mechanisms shall provide variable heat rejection per the requirements of the temperature control subsystem.

d. Star Tracker Latch

The star tracker latch shall restrain the tracker in the launch position. Upon command, the latch shall release, allowing the tracker to be positioned as required.

10. Thermal Control Subsystem (TC)

a. Function

The function of the thermal control subsystem is to maintain the temperatures of all space vehicle elements within their design temperature ranges for the duration of the Encke Rendezvous mission. The design of the TC subsystem takes into account the various duty cycles and changes in the external thermal environment to which the space vehicle will be exposed

Unlike other space vehicles designed to operate in a solar environment of less than 1 earth sun (e.g., Viking '75), or those designed to operate at more than 1 earth sun (e.g., Mariner Venus-Mercury '73), the Encke Rendezvous space vehicle must be capable of performing at both extremes. Furthermore, the extremes themselves are far more severe than those experienced by the two examples cited. In fact, the wide variation in solar irradiance during the mission lifetime is the primary consideration in the design of the thermal control subsystem.

b. Composition

This section describes the elements of the space vehicle thermal control subsystem. These are:

- (1) Thermally actuated louver assemblies
- (2) Thermal insulation blankets
- (3) Thermal shields

The following elements (4, 5, and 6) are not ordinarily considered to be part of the thermal control subsystem, however, they are included for the sake of completeness:

- (4) Thermal control coatings and surface finishes
- (5) Electrical heaters
- (6) Specially designed conductive paths.

c. Implementation

The thermal control of the space vehicle is implemented from the element to the system level, since the temperature control of the various subsystems are not, in general, independent of one another. Consequently, it is impossible to treat the spacecraft and the SEP module as independent thermal entities.

d. Functional Descriptions

1) Thermally Actuated Louver Assemblies. Effective emittance control louver assemblies shall be used to provide variable heat rejection capabilities from the radiating surfaces of the spacecraft electronic bays.

2) Thermal Insulation Blankets. Thermal insulation blankets will be used wherever necessary to limit environmental heat exchange and heat exchange between spacecraft elements.

3) Thermal Shields. Thermal shields will be used where necessary to provide sun shading or to effect infrared blockage between spacecraft elements.

4) Thermal Control Coatings and Surface Finishes. Coatings will generally be in the form of a paint or a plating; surface finishes will usually be polished, oxidized, or otherwise treated metallic surfaces.

5) Electrical Heaters. These will be electrically purely resistive impedances which provide Joulean heating.

6) Specially Designed Conductive Paths. These will usually, but not always, be low conduction paths where thermal isolation is required.

#### 11. Science Instruments

Table II-E-2 lists the instruments which are proposed as the minimum "fundamental" payload for an Encke rendezvous by the 1972 TRW Encke rendezvous study.<sup>1</sup> Since this study is the most recent and extensive in the area of cometary payload selection, the recommended payload has been tentatively selected for this spacecraft. However, it must be recognized that no detailed studies of individual instruments were made in order to guarantee that each would actually operate properly in and on the cometary environment, and no instrument was included capable of determining solid particle composition. Some change in this payload seems inevitable when it receives a more detailed analysis. As the table shows, most of the demanding instruments have application to properties of both the coma and nucleus.

The list of instruments splits into two groups, those mounted in a fixed position and orientation, and those requiring a scanning capability. The fixed group consists of the optical particle detector, magnetometer, plasma

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<sup>1</sup> TRW Systems Group, Study of a Comet Rendezvous Mission, Final Technical Report, Volume I, May 12, 1972, p. 4-3.

Table II-E-2. Science Payload Complement for an Encke Rendezvous Mission

Instrument	Property to Which Applied
TV Image (100 rad resolution)	Size of nucleus Rotation of nucleus Shape of nucleus Appearance of details of nucleus Size of halo Shape of halo Size of coma Shape of coma Size of tail (uncertain) Shape of tail (uncertain)
Multichannel White Light Photometer	Albedo of nucleus Phase function of nucleus Albedo of halo Phase function of halo Brightness profile of halo
IR Radiometer	Temperature of nucleus
Photopolarimeter	Fine scale texture of nucleus Fine scale size distribution of ice grains of halo Fine scale size distribution of nonvolatile particles of coma
Microwave Altimeter	Mass of nucleus Size of nucleus Surface composition of nucleus
Radiometer, UV 1000-4500Å	Distribution of ionized gases in coma, contact surface, and tail
Optical Particle Detector (Sisyphus)	Distribution, velocity of icy grains of coma Distribution, velocity of nonvolatile particles of coma
Mass Spectrometer	Flux, velocity, density, spatial distribution of neutral and ionized gases of coma
Magnetometer	Magnetization of nucleus Magnetic field configurations of contact surface, tail, and interaction region
Plasma Wave Detector	Electric waves in contact surface, tail, and interaction region Local electron densities in ionized coma
Langmuir Probe	Local electron densities in ionized coma
Plasma Probe	Flux, density, energy spectrum of solar wind and reduced solar wind in interaction region

wave detector, and Langmuir probe. The scanning group includes detectors that will predominantly follow the nucleus: the TV camera, photometer, IR radiometer, mass spectrometer, and microwave altimeter; and detectors that will scan the coma or interaction region as well, e.g., the UV radiometer, photopolarimeter, mass spectrometer, and plasma probe.



## F. SEP MODULE SUBSYSTEM DESCRIPTIONS

### 1. Thrust Subsystem (TSS)

The function of the thrust subsystem (TSS) is to provide the directed impulse required to accomplish a rendezvous with the comet P-Encke ( $\sim 1.5 \times 10^7$  newton-sec) and, while operating, to provide control torques for vehicle three-axis attitude stabilization. To provide the impulse, the TSS must convert solar-array-derived electrical energy into the directed kinetic energy of ejected propellant (mercury) for a period of approximately 950 days. The primary elements comprising the TSS are:

- (a) Thruster
- (b) Power conditioner (PC)
- (c) Switching matrix
- (d) Thrust vector control (TVC)
- (e) Propellant tankage and distribution
- (f) Cabling
- (g) Structure.

The functional relationships of these elements are shown in Fig. II-F-1 and are described below. (See functional description of the thrust subsystem, Appendix A, for a more detailed description.)

#### a. Thruster

Seven thrusters (this number of thrusters contains excess thrust capacity), producing thrust according to a preset schedule, deliver the needed total impulse of  $1.5 \times 10^7$  newton-sec at a specific impulse of 3000 sec. Each thruster is a 30-cm diameter configuration and has structural mounting pads to which the gimbal motor shafts can be attached. Propellant from the propellant tankage and distribution element is manifolded to each of three vaporizers: hollow cathode, main, and neutralizer.

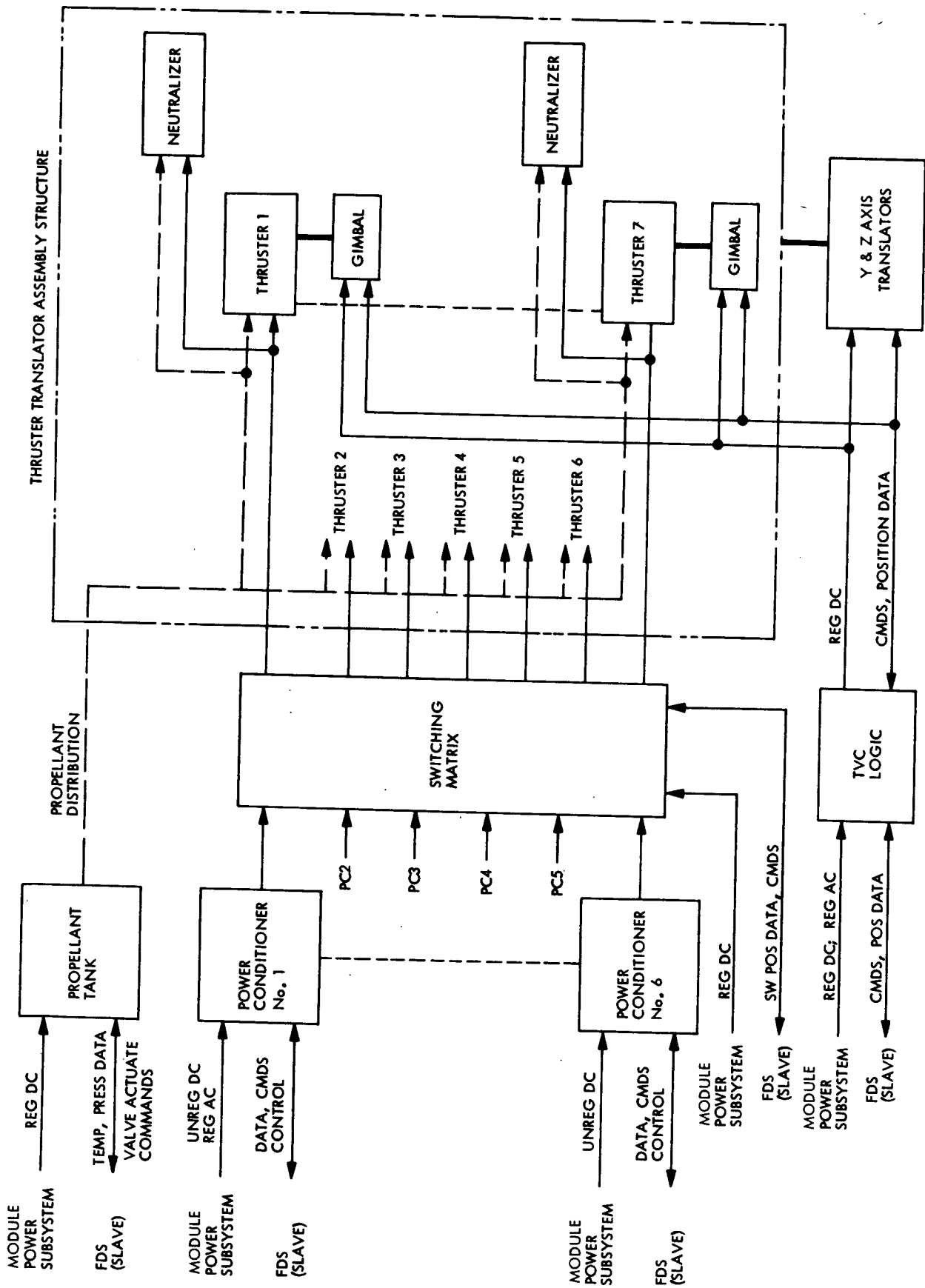


Fig. II-F-1. Thrust Subsystem Functional Block Diagram

b. Power Conditioner (PC)

The function of the PC is to convert approximately 2860 watts of unregulated dc power supplied by the solar arrays to the various ac and dc powers required to operate the thrusters, and to provide control of thruster operation. Each of the PC's has 11 outputs which are connected through switches (in the switching matrix) to the seven thrusters. Power with a 2.4 kHz, 50-Volt rms square wave voltage waveform (from the power subsystem) is used for internal PC power supply turn-on operation.

Control loops within the PC function to control the output of the thruster that is providing power by closely regulating the thruster output, using an externally provided reference signal. In addition to being self-protected from any short circuits caused by thruster operation, each PC contains the necessary circuitry to suppress thruster arcing and to automatically restore thruster operation after arcing has been suppressed.

Each PC is individually controlled by digital commands, as required for turning the unit on and off. Voltage, current, and temperature signals are used by the flight data subsystem to monitor PC operation.

c. Switching Matrix

The switching matrix is used to switch the electrical connections of six power conditioners to seven ion thrusters. The switching matrix comprises:

- 1) Six rotary switches, each consisting of five decks (four input contacts per deck), which switch power conditioner outputs to any thruster, and
- 2) The switching matrix assembly which provides switch mounting, interconnection of switch outputs, and interfaces to the power conditioners and thrusters.

Auxiliary power delivered at regulated dc voltage is used for actuating the switches on command from the FDS.

d. Thrust Vector Control (TVC)

The TVC provides control torques about the three primary spacecraft axes utilizing the thrust from the thrusters. This is accomplished by translating the average thrust vector of all thrusters parallel to two axes through the spacecraft center of mass, thereby providing pitch and yaw axis control, and by gimbaling individual thrusters to provide a couple about the third axis. The TVC operates in closed loop with the vehicle attitude control star tracker and sun sensors. Vehicle attitude as sensed by the star tracker and sun sensors and vehicle rate determined by the rate estimator will be used to command repositioning of the thruster translator assembly and gimbaling of thruster pairs. Proportional control of stepper motor rates will be used for control about any axis proportional to the error in that axis.

The TVC consists of:

- (1) Six gimbal actuators (the center thruster is not gimballed)
- (2) The translator carriage
- (3) Translator rods
- (4) Two translator actuators
- (5) TVC logic.

Auxiliary power delivered at a regulated dc voltage operates the translator and gimbal actuators. The TVC logic operates from regulated ac at 50 volt rms, 2.4 kHz square wave.

During the mission the TVC operates at all times that thrusting is taking place. The spacecraft reaction control system (RCS) is activated if the deadband limits should be exceeded. Once the position and rates again fall within the acceptable limits, TVC operation is automatically resumed.

e. Propellant Tankage and Distribution

The propellant tankage and distribution element:

- (1) Contains the thruster's propellant pressurant storage reservoirs
- (2) Delivers mercury propellant to thrusters in required pressure range.

Pressurization is accomplished by utilizing the vapor pressure of Freon TF.

The principal components of the element are:

- (1) A tank with an expulsion bladder (diaphragm)
- (2) Pressure and temperature transducers
- (3) A tank valve
- (4) The propellant distribution network (including, as required, the feedlines, flow distributors, manifolds, and any shutoff valves which might be required near the thrusters to prevent propellant expulsion at a malfunctioned vaporizer).

Heaters around the tank would be required if the local environment is not adequate for maintaining the Freon temperature at the level required for pressurization. In addition, thermal control of portions of the distribution network could be required to maintain mercury in its liquid state throughout the mission.

A single spherical tank (0.4 m dia) would be required for storing the required 480 kg (1056 lb) mercury propellant. The tank valve (solenoid-actuating type) prevents propellant exiting the tank at launch, and then allows propellant to exit from the tank to the thrusters when required. Regulated dc power actuates the tank valve upon command from the FDS. Operation of the element is monitored through pressure and temperature data transmitted to the FDS.

f. Cabling

TSS cabling includes all cabling required to make electrical connections to the various components and independently-mounted units. It includes:

- (1) A power conditioner compartment RING harness (PC RING) for interconnecting the PC's to the switching matrix assembly and to the power subsystem, as well as incorporating the connectors and cables interfacing with the propellant tank, and other SEP module electronics
- (2) A power conditioner assembly harness interconnecting the PC to the PC RING,
- (3) The thruster array harness to provide the interconnection of electronic equipment mounted on the thruster translator assembly structure and the interface to the switching matrix assembly.
- (4) The pyrotechnic harnesses for actuation of electro-explosive devices and solenoid valves serving such functions as unlatching translating caging devices, and shutting off mercury flow to a malfunctioning thruster.

g. Structure

The TSS structure includes:

- (1) Thruster structure
- (2) Power conditioners structure,
- (3) Thruster translator assembly structure
- (4) Propellant tank support structure.

The function and description of each of these are as follows:

- 1) Thruster Structure. The thruster structure structurally integrates each thruster's components into its configuration. It shall provide

mounting shafts or flanges to which shafts can be mounted, in order that proper interfacing with the TVC gimbal actuators can be accomplished.

2) Power Conditioner Structure. The PC structure is primarily in the form of module plates. This structure packages the PC components, and also serves as shear plates for the spacecraft structure.

3) Thruster Translator Assembly (TTA) Structure. The TTA is defined as that assembly which includes seven thrusters, TVC components (less the TVC Logic), cable troughs for carrying feedlines and cables across articulating interfaces, feedlines and cables, any thermal control items and the TTA structure. The function of the TTA structure is to integrate these components and elements into an assembly, and the assembly to the spacecraft; it includes latches that secure the TTA to the spacecraft structure in such a way during launch that lateral and axial motion during this event is prevented.

4) Propellant Tank Structure. The propellant tank structure integrates the propellant tank into the spacecraft.

## 2. SEP Module Power Subsystem (MPSS)

The power subsystem of the SEP module generates, processes and distributes unregulated and regulated power for use by the:

- (a) Thrust subsystem and other SEP module subsystems
- (b) Spacecraft power subsystem

A functional block diagram of the power subsystem is shown in Figure II-F-2. The MPSS is composed of the following major elements:

### a. Solar Array

The primary power source consists of two solar arrays of the roll-up configuration. The basic roll-up array design, which has evolved from a JPL-sponsored development program, has a Kapton membrane which serves

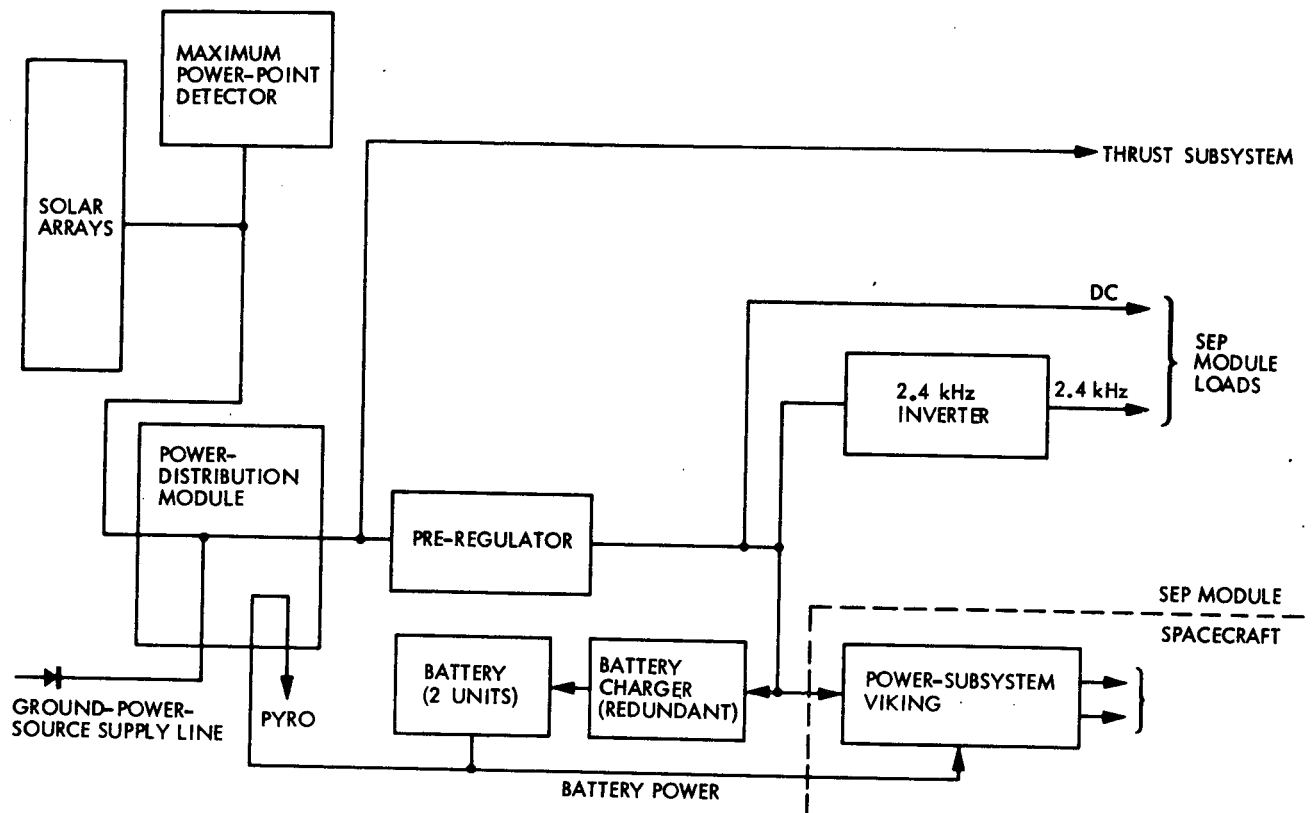


Fig. II-F-2. SEP Module Power Subsystem with Mariner/Viking Spacecraft

as a flexible substrate upon which the solar cells are mounted. The flexible substrate is rolled onto a drum in a manner similar to a window shade, for storage during launch. Deployment of the substrate is achieved via an extensible motor-driven boom which also provides the required structure (aided by a leading edge member) to maintain the flexible substrate in a planar configuration shown in Fig. II-F-3. A roll-up array has a nominal power producing capability (at earth) of  $107.6 \text{ W/m}^2$  (10 Watts per square foot) at a temperature of  $60^\circ \text{C}$  and a solar intensity of  $140 \text{ mw/cm}^2$  with the array normal to the sun. To generate 20 kW at 1 AU, an area of  $186 \text{ m}^2$  ( $2000 \text{ ft}^2$ ) total is required. Two solar arrays, each having an area of  $93 \text{ m}^2$  ( $1000 \text{ ft}^2$ ) will be used. Typical dimensions for each array are 4.3 meters wide by 22.8 meters long (14 ft by 75 ft). The specific power density is expected to be approximately  $66 \text{ W/kg}$  ( $30 \text{ W/lb}$ ). Temperature, intensity and the sun angle of incidence combine to influence the power-producing capability of the array. For example, for an Encke comet mission extending from 1.0 AU to 3.3 AU, the power output of the



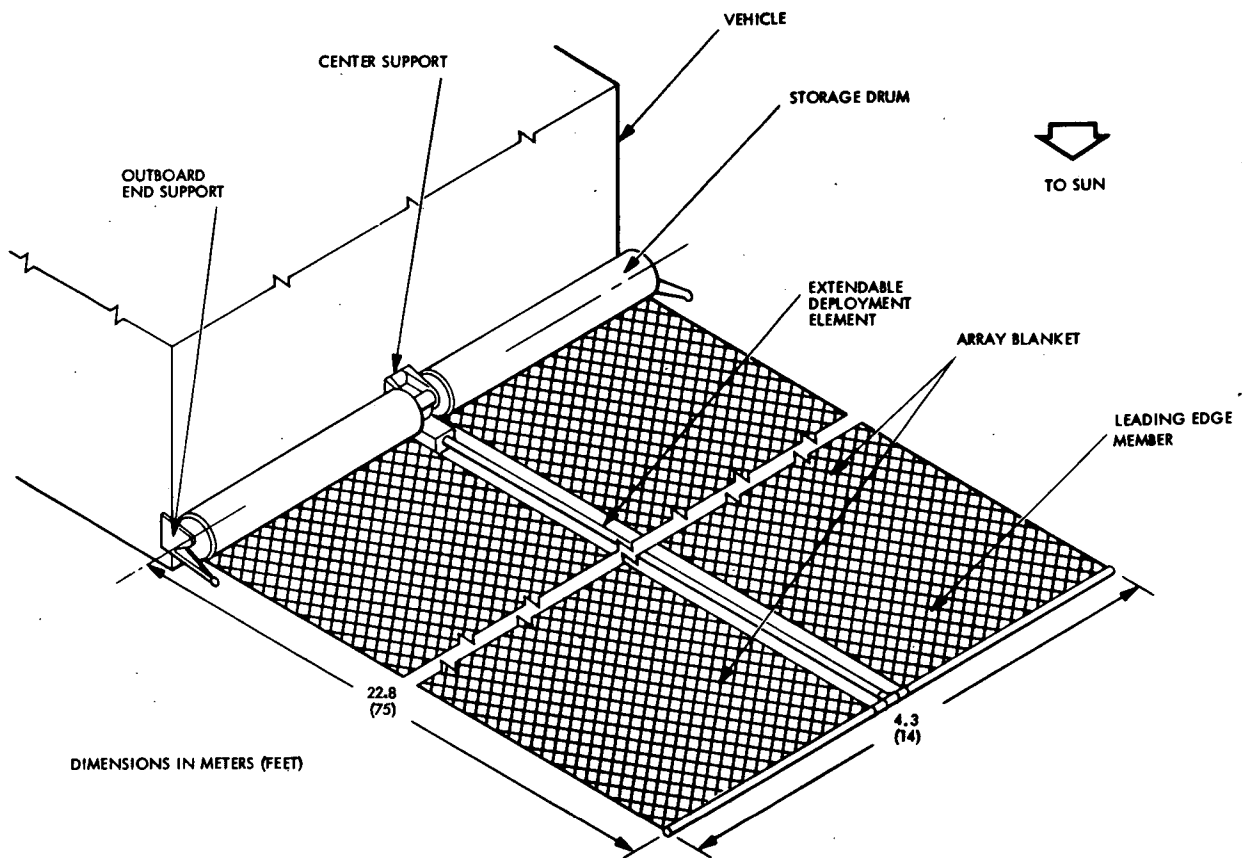


Fig. II-F-3. Solar Array Configuration

Array will vary from 20 kW to 2.65 kW. The output voltage of the array will vary from 200 Volts up to approximately 400 Volts.

b. Maximum Power Point Detector

The maximum power point detector determines the maximum power of the array source upon command from the central computer. It also measures the power required by the spacecraft and determines the power margin available (the difference between maximum power and power consumed).

c. Power Distribution Module (PDM)

The power distribution module receives all of the solar array power, which is then redistributed to:

- (1) The thrust subsystem, where the power conditioners condition the power for use by the ion thrusters
- (2) The SEP module power conditioning, which utilizes a pre-regulator to condition the power for the SEP module dc and 2.4 kHz loads, and for the spacecraft engineering subsystems and science instruments, via the spacecraft power subsystem.

Ground power is supplied to the power distribution module for operation of the thrust subsystem and SEP module subsystems during system tests. Power is supplied to the power distribution module from the battery source for firing the solar array release squibs after liftoff and deploying the solar arrays.

The power distribution module contains the bus bars and junction points of the power subsystem power cables, in addition to the telemetry sensors required for evaluation of the power subsystem performance.

d. Pre-Regulator

The pre-regulator accepts power from the power distribution module and generates the necessary voltage for the SEP module inverter, SEP module dc loads and the spacecraft power subsystem. The pre-regulator output voltage must be compatible with the existing Viking power subsystem equipment in order to avoid design changes in the latter. An output voltage between 40 and 50 Volts is required to ensure battery charging and proper operation of the Viking booster regulator. In addition, the pre-regulator will be utilized as

a filter between the thrust subsystem and the spacecraft to reduce noise and transients to an acceptable level for the spacecraft subsystems.

e. SEP Module Power Inverter

The propulsion housekeeping power inverter is driven by the pre-regulator and delivers a 2.4 kHz, 50 Volt rms squarewave. The 2.4 kHz frequency was selected to utilize existing Viking designs and hardware.

f. Battery

The battery has a 30 ampere-hour capacity rating. The Viking spacecraft has two-30 AH batteries to supply the orbit insertion and orbit requirements. One battery will be sufficient to supply the power requirements envisioned for the mission proposed.

g. Battery Charger

The battery charger provides dc power to charge the battery, after an energy discharge cycle, and maintains the battery in a fully charged state. The battery charger is a relatively new design providing three charge rates (high rate, medium rate, and low rate). The rate selected is dependent upon battery state of charge and the power available. The charge rates are controlled by commands from the MFDS.

3. Master Flight Data Subsystem (MFDS)

The main functions of the Master FDS, as illustrated in Fig. II-F-4, are the following:

- (a) Accepts coded commands from the CCS and stores/processes them.

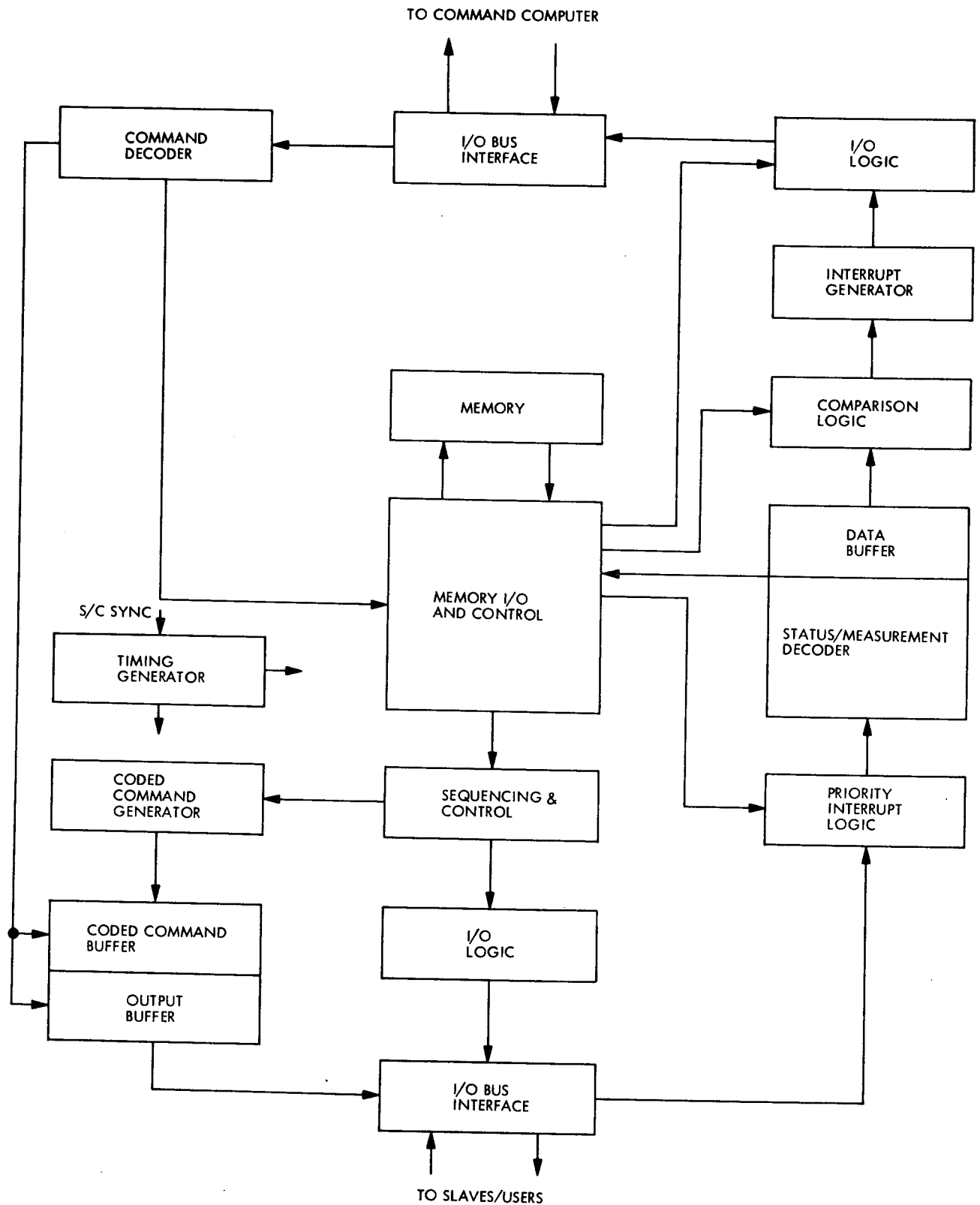


Fig. II-F-4. Master Flight Data Subsystem Functional Block Diagram

- (b) Sends routine commands to subsystem slaves on a cyclic, fixed format basis to accommodate unchanging requirements. Refreshes data on a cyclic basis.
- (c) Sends special commands to subsystem slaves as generated by the CCS, to take corrective action, accommodate failures, meet new requirements, etc. Refresh may be provided.
- (d) Receives status and measurement information from subsystem slaves on a priority basis.
- (e) Compares status and measurement information with stored or preset values and generates interrupts or flags for the CCS.
- (f) Transfers out-of-tolerance or selected measurements to the spacecraft FDS under program control.

Whenever there is a need for minimum response time to CCS commands by a SEP module subsystem, the sequencing and control logic may be bypassed. A command could be routed directly from the command decoder to the output buffer. The coded command buffer would be inhibited from transferring commands from the sequencing logic and the command transferred immediately to the appropriate subsystem.

To respond efficiently to the recurrent needs of the subsystems, as well as exceptions caused by faults, failures and out-of-tolerance measurements, the Master FDS should have a priority interrupt capability for selecting the most important input to be processed next. Very careful consideration should be given to the priority interrupt mechanization. The scheme should allow for queuing of successive interrupts such that no user/slave goes unrecognized. Also, provision should be made for masking the normal order of priorities. Interrupt masks would be generated by the CCS.

The Master FDS should be able to compare measurements and status against known limits and selected operating states. It should also be

possible to alter these standards by using stored information or commands from the earth. To complete the local control loop, interrupts or flags could be sent to the CCS whenever out-of-tolerance measurements or unacceptable states are detected. The CCS would respond with corrective action.

The memory should be capable of storing the following types of information:

- (a) Commands
- (b) Status/measurement data
- (c) Tolerance levels and status bit patterns for the comparison logic
- (d) Priority interrupt masks.

It should be word length compatible with the spacecraft FDS and the CCS on a one-to-one or multiple word basis.

As already established by the overall system constraints, the FDS Master will facilitate the closed-loop operation of the SEP module. Transfer of data to the spacecraft FDS should be on an exception or as-requested basis.

#### 4. SEP Module Structure Subsystem (MSTRU)

The SEP module structure subsystem consists of three primary elements; the power conditioner compartment, the SEP electronic compartment and the solar array support structure. The main structural members of these elements are indicated in Fig. II-F-5.

The power conditioner compartment is the largest structural element of the SEP module. It contains the interface with the launch-vehicle adapter and thus supports all other spacecraft structural elements. The structure of the PC compartment consists of eight longerons which connect upper and lower frames and provide support for the six power conditioner units which mount on opposite sides of the PC compartment. The outer surfaces of the

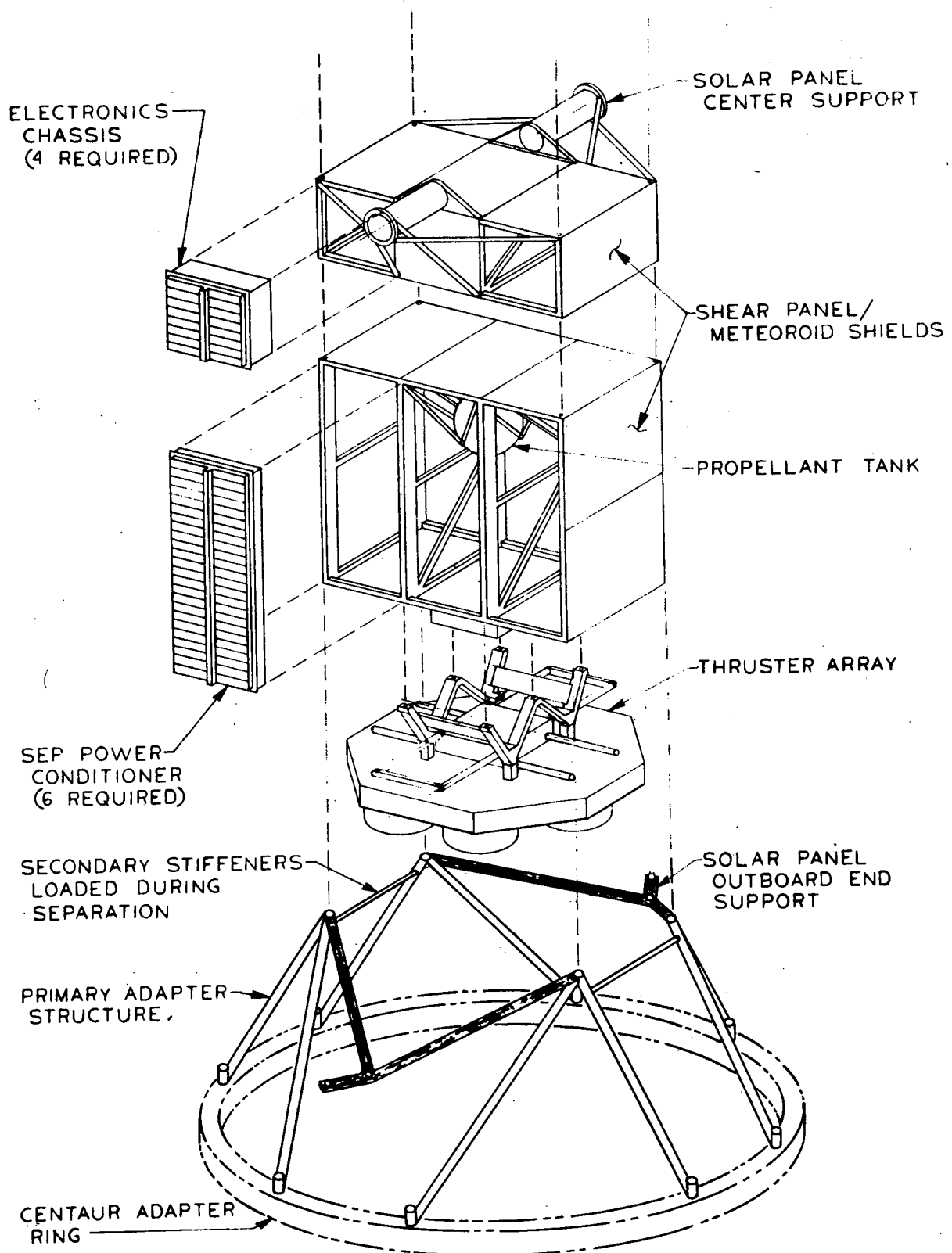


Fig. II-F-5. SEP Module Structure

power conditioner units serve as shear plates in the structure and also provide meteoroid protection when used in conjunction with the PC louver assemblies. The remaining external surfaces of the PC compartment are stiffened by shear panels and shear panel intercostals. These surfaces are also required to serve as meteoroid shields.

The internal longerons are tied together with intercostals and diagonal bracing to form two deep beams which run the length of the PC compartment. These beams, together with the narrow sides of the compartment provide the rigidity required by the power conditioning units. The beams also support the mercury propellant tank via an eight member truss.

The bottom frame of the PC compartment provides the interface with the launch-vehicle adapter structure and supports the SEP thruster translator assembly and switching matrix. The upper frame provides the separable interface with the SEP electronics compartment.

The SEP electronics compartment is constructed similarly to the PC Compartment except that the PC units are replaced with electronic chassis. The bottom frame of the electronic compartment provides the interface with the PC compartment and the top frame provides the separable interface with the spacecraft supporting truss. The electronic compartment also provides the primary structural interface with the solar array support structure.

The solar array support structure is the third major structural element of the SEP module. As currently configured the General Electric/JPL rollup solar array design requires a primary center support and two secondary outboard end supports which are used only during launch. The primary center support is supported from the SEP electronics module by a center support tube and associated braces. To minimize spacecraft weight, the structure required to support the lower ends of the arrays during launch is attached to the launch-vehicle adapter, and remains with the launch vehicle following separation.

Though the current solar array design also requires an upper outboard end support, an analysis of the solar array interface requirements indicates that the upper supports can be removed in future designs. The upper



outboard end supports have therefore not been included. Their removal greatly simplifies the spacecraft configuration and improves its multi-mission capability by not requiring a solar panel structural interface above the SEP module.

5. Attitude Control Subsystem (TVC)

The attitude control subsystem for the SEP module is called thrust vector control, and denoted TVC. The functions of the TVC are:

- (a) To maintain desired attitude of the space vehicle while thrusting is taking place. Position and rate information is used to produce counteracting torques by translating the thruster array and/or gimbaling the thrusters.
- (b) To change the desired attitude of the space vehicle (i.e., thrust vector pointing) by mechanically gimbaling the star tracker and rotating the solar arrays such that the space vehicle attitude changes in response to the apparent celestial sensor errors. Solar arrays are normally maintained perpendicular to the sun by means of solar array rotational actuators. A second degree of freedom with respect to the space vehicle is obtained by gimbaling the sun sensors mounted on the solar arrays, allowing the arrays to point off the sun if necessary.

Thrust vector control is obtained by translating the thruster array parallel to the roll (z) and yaw (y) axes, and by gimbaling thrusters in pairs. Translating the thruster array produces unbalanced torques on the vehicle about the roll and yaw axes. These torques are used to counter disturbance torques from solar pressure, mass center offset, etc. Gimbaling two thrusters symmetrically placed about the geometric center of the thruster array produces a control torque for the pitch (x) axis.

The SEP module components of the TVC are described as follows:

a. Thrust Vector Control Electronics

The electronics units required to control a single axis of the space vehicle are:

- (1) Voltage controlled oscillator (VCO).
- (2) Bi-directional pulse generator.
- (3) Stepper motor drive and logic.
- (4) Compensation network.
- (5) Rate and position estimator.

Three subsets of units are required for 3-axis control of translator and gimbal positions. All logic for the TVC, except for logic contained in the star tracker, is in the TVC electronics.

b. Thruster Array Translational Actuators

The translational actuators are located in the thrust subsystem. Two actuators are used, one for each axis. Each actuator consists of a

- (1) Size 15, 90 degree permanent magnet stepper motor.
- (2) Conventional spur reducing gear.
- (3) Harmonic drive reducing gear.
- (4) An infinite-resolution, conducting plastic, single-turn potentiometer.

The potentiometer monitors output shaft position. The rotating output shaft is keyed to a drum and strap drive to provide the required linear motion of the thruster array parallel to either the yaw or roll axis.

c. Thruster Gimbal Actuators

The thruster gimbal actuators are located in the thrust subsystem. One is used for each gimballed thruster. A total of six are required, as the seventh (center) thruster is not gimballed.

Each actuator consists of a size 11, 90 degree permanent magnet stepper motor, strap drive, reduction gearing, including a worm gear stage to prevent back-driving, and an output position feedback linear variable differential transformer. One step of the stepper motor produces a 0.1 mrad rotation of the output shaft which controls gimballed thruster orientation.

d. Solar Array Mounted Sun Sensor

The solar array mounted sun sensor consists of photoconductive cells and associated electrical circuits. The assembly provides the sensing function for yaw attitude control of the space vehicle. In addition to the inherent gimbaling provided by the rotatable solar array, the sensor is gimballed to allow pointing the panels off the perpendicular to the sun.

e. Solar Array Rotation Actuators

Two solar array rotation actuators are used to rotate the arrays about the space vehicle yaw axis. In addition, TVC uses the spacecraft mounted sun sensor and star tracker. Figures F-6 and F-7 are block diagrams for the pitch, roll and yaw axes while under TVC.

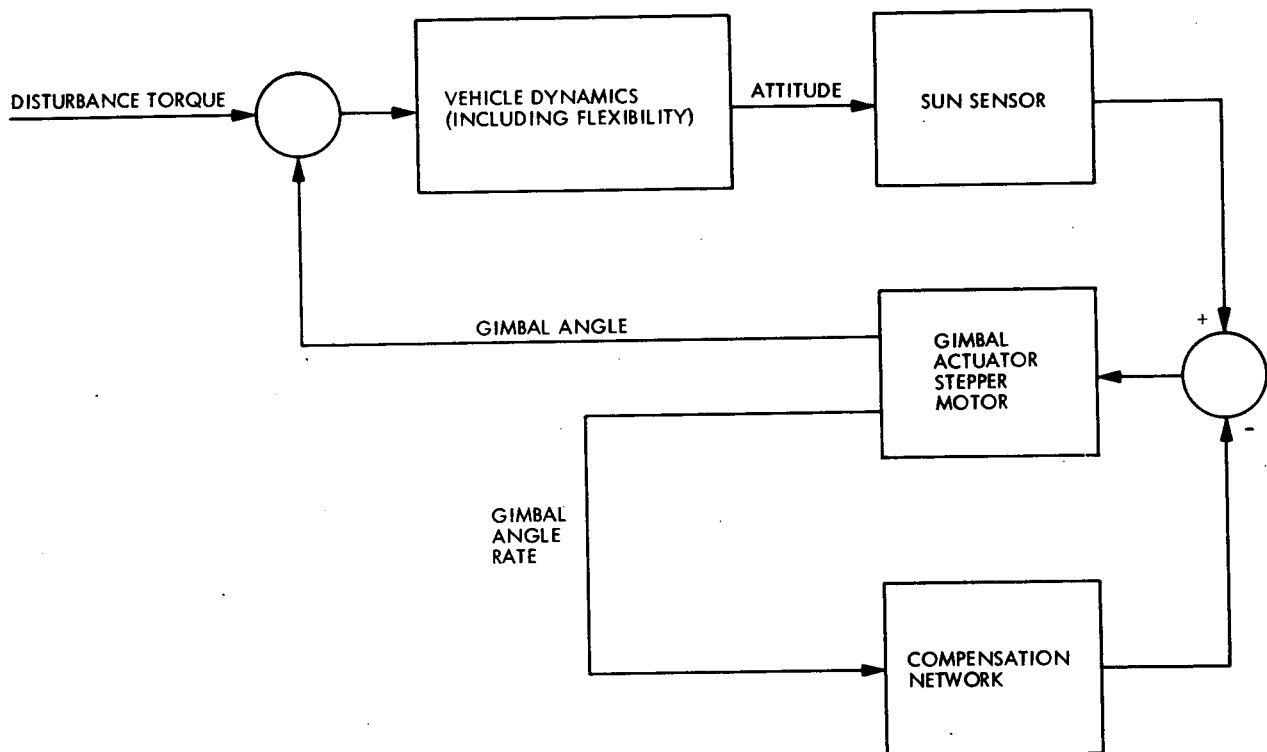


Fig. II-F-6. Pitch Axis Conceptual Block Diagram

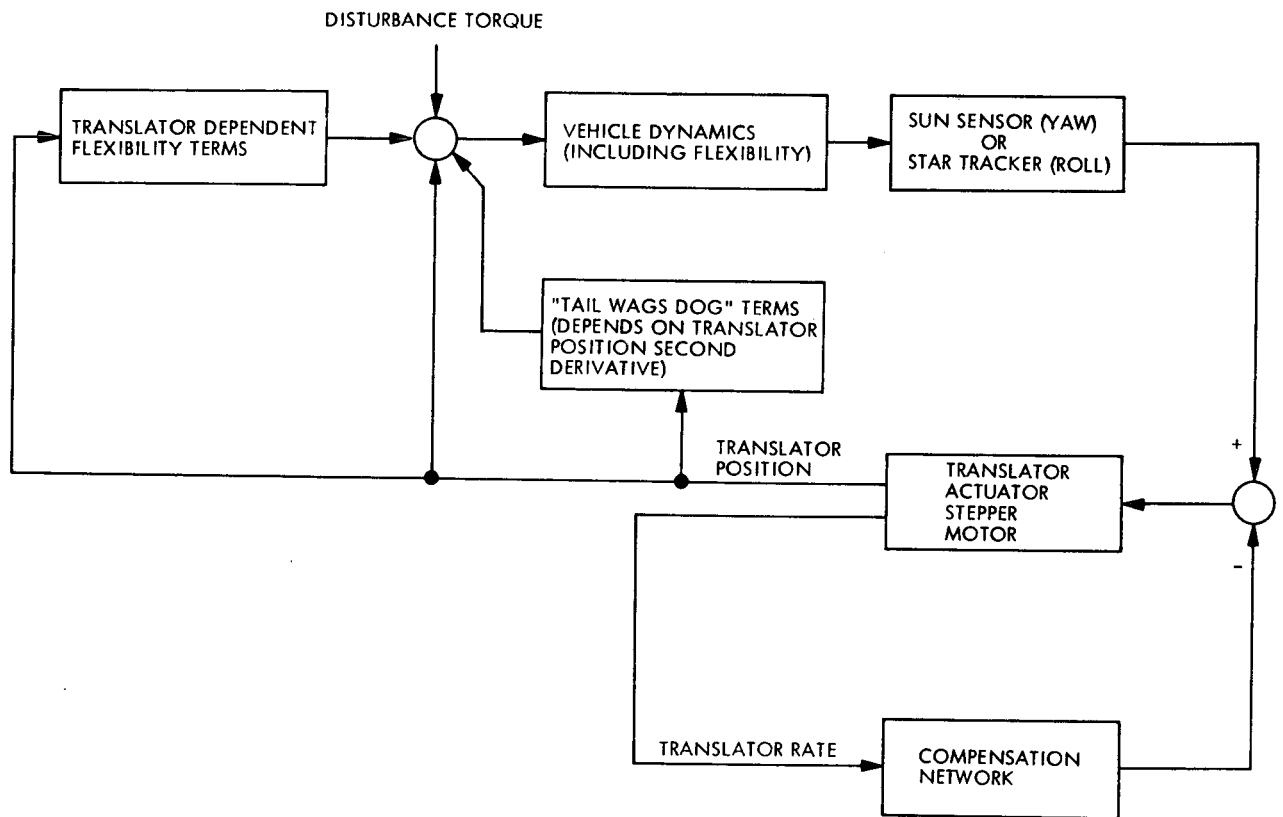


Fig. II-F-7. Roll or Yaw Axis Conceptual Block Diagram

6. SEP Module Cabling Subsystem (MCABL)

a. Scope

The SEP module flight cabling subsystem provides the electrical interconnections between flight subassemblies, assemblies, subsystems and for the required interfaces to the solar array, thruster array, flight spacecraft and to the launch support equipment. All cabling required to make these electrical connections on the SEP module and to adapt it to the spacecraft are included in the cabling subsystem. Not included is wiring that is part of subassemblies or any rf coaxial cabling.

b. Design

Cabling design assumes maximum use of VO'75 design and technology. Major design features include the use of #26 AWG copper alloy conductor with TFE insulation for signal circuits and #22 or #12 AWG for power circuits. Rectangular Micro-D Series or Circular "JT" type connectors will be used for harness interfaces.

Flight electrical cabling is designed to:

- (1) Transmit the signals and power in a manner compatible with the requirements of the source unit and the destination unit.
- (2) Control induced electrical interference.
- (3) Control power loss.
- (4) Control voltage drop.
- (5) Shield wires carrying otherwise incompatible signals or power.
- (6) Provide adequate electrical insulation.
- (7) Provide convenient electrical connect/disconnect capability for installation and removal of each subsystem.

c. Description

The SEP module cabling subsystem will include approximately 28 cables. These cables are of five major types, as described below:

1) Support Electronics Assembly Harnesses

Four harnesses interconnect the subassemblies of the SEP module support electronics subsystems and provide connections for system level functions to the system RING harness in the electronics bay. Connectors for "direct access" of test functions are provided on one end of the bracket supporting the harness assembly, if required by the subsystem.

2) System RING Harnesses

Four RING harnesses are contained within the support structures for the support electronics compartment and the power conditioner compartment. In the electronics compartment, the upper RING harness is contained within a support structure located within the primary structure. To control cross coupling, the design of the upper RING harness provides for physical separation of wire groups and, wherever possible, the associated connectors for power, noisy-signal, and quiet-signal circuits. The connectors of the upper RING harness are either hard-mounted to the supporting structure or connected to hard-mounted connectors on the electronic assemblies. The connectors are accessible from the sides of the compartment (with thermal shield removed).

If required, a lower RING harness will be provided on the compartment structure adjacent to the lower periphery of the electronic assemblies. Cable branches out of the lower RING harness interconnect the electronic assemblies with the spacecraft interface cabling, the solar array cabling, and the power conditioner compartment RING harnesses.

Two RING harnesses are located at opposite ends of the power conditioner compartment and serve to carry power and signal circuits within the compartment. The upper RING contains the interface of the power distribution subsystem and the support electronics to the individual power conditioner inputs. Some system level circuits are carried by cabling between the power conditioner compartment upper and lower RINGS, however, principal power flow is through the power conditioner assembly harnesses and then into the lower RING harness.

The power conditioner compartment lower RING harness contains the interface connections to the switching matrix assembly from each power conditioner and the systems interface connectors to the thruster array.

3) Power Conditioner Assembly Harnesses

Each power conditioner assembly is provided with a harness to interconnect the modules of the assembly. The input connectors and output connectors are located at opposite ends of the assembly and cabling routing and support is integrated with the power conditioner structure. Connectors are hardmounted on the structure to accept plug-in modules. To control EMI, and minimize cabling weight, cables are bundled and routed by wire groups based on power, noisy-signal, and quiet-signal circuit requirements on opposite sides of the central support for the module attachment. Hardmounted circular connectors interface with the power conditioner compartment RING harnesses.

4) Thruster Array Harnesses

Six specially designed ribbon cable harnesses are provided on the thruster array. These harnesses provide the interconnection from the power conditioner compartment lower RING harness and from the switching matrix assembly to the various assemblies, devices, and electrical systems mounted on the array. In addition to maintaining separation of power, signal and high-voltage circuits, the cabling system provides the mechanization of the cable routing in support troughs which provide a rolling articulation of the cables across the array translation axes. The cables separately handle the power and control inputs/outputs to the thrusters, gimbal actuators, and translator actuators. The cables are maintained and routed in the ribbon configuration, breaking out to round cable bundling just before reaching the interface connectors.

5) Additional SEP Module Cables

Additional electrical cables are supplied as necessary for equipment not contained within the SEP module compartments or on the thruster array. Such cabling includes pyrotechnic distribution lines, attitude control gas-jet wiring, temperature transducer cabling, solar array input cabling to the support electronics RING interface, and intersystem cabling to the

spacecraft. The physical and electrical characteristics of these special harnesses accommodate any special requirements of the equipment to be interconnected.

7. SEP Module Mechanical Devices Subsystem (MDEV)

The mechanical devices subsystem for the SEP module consists of the following mechanisms:

a. Thruster Array Latches

The thruster array latches shall restrain the thruster array to the primary structure. The thruster array latches shall prevent launch loads from feeding into the array positioners and bearings. Upon command, the latches shall release and not interfere with array motion.

b. Solar Array Latches and Positioners

The solar array latches shall restrain the arrays to the spacecraft structure in a launch position. Upon command, the latches shall release and allow the arrays to individually rotate upon command. The positioning mechanisms shall rotate each array at a controlled rate such that the pointing vectors of each are parallel and in the same direction.

c. Space Vehicle Latches

The space vehicle latches shall restrain the space vehicle to the adapter. Actuation of the latches shall result in release of the space vehicle in such a manner that will ensure unobstructed separation of the space vehicle and minimize post-separation debris.



d. Separation Mechanisms

The space vehicle separation mechanisms shall be mounted in the space vehicle adapter at alternate corners. The mechanisms will most likely consist of four spring-loaded pistons that will be preloaded against pads on the space vehicle lower structural surface. The mechanisms shall provide separation that will result in a separation velocity and tip-off rate consistent with the space vehicle inertial properties specifications.

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APPENDICES

- A. THRUST SUBSYSTEM DETAILED FUNCTIONAL DESCRIPTION
- B. DESIGN REQUIREMENT, THRUST SUBSYSTEM  
FLIGHT ELECTRONIC PACKAGING
- C. FLIGHT EQUIPMENT, THRUST SUBSYSTEM CABLING  
AND CABLING HARNESES

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## APPENDIX A

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### THRUST SUBSYSTEM DETAILED FUNCTIONAL DESCRIPTION

#### TABLE OF CONTENTS

1.0	SCOPE
2.0	APPLICABLE DOCUMENTS
3.0	FUNCTIONAL DESCRIPTION
3.1	Function
3.2	Requirements
3.3	Subsystem Elements
4.0	INTERFACES
4.1	SEP Module Power Subsystem Interfaces
4.2	SEP Module Structure Interface
4.3	Data Subsystem Interface
4.4	Attitude Control Subsystem Interface
4.5	SEP Module Thermal Control Subsystem Interface
4.6	SEP Module Cabling Subsystem Interface
5.0	PERFORMANCE PARAMETERS
6.0	PHYSICAL CHARACTERISTICS AND CONSTRAINTS
7.0	SAFETY

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## APPENDIX A

### THRUST SUBSYSTEM DETAILED FUNCTIONAL DESCRIPTION

#### 1.0 SCOPE

This functional description covers the subsystem description, system requirements, and interface requirements of the electric thrust subsystem for the Encke 1980 Rendezvous Mission.

#### 2.0 APPLICABLE DOCUMENTS

The following documents form a part of this functional description.

##### REQUIREMENTS

Jet Propulsion Laboratory

TSS-220-1*	Design Requirement, Thrust Subsystem Flight Electronic Packaging
TSS-2009-1*	Design Requirement, Flight Equipment Thrust Subsystem Cabling, Cable and Harness Assemblies

##### SPECIFICATIONS

Jet Propulsion Laboratory

ZPP-2063-PMP*	Preferred Materials, Fasteners, Processes, and Packaging and Cabling Hardware
DM 505139*	High Voltage Protection
CS 506216, Rev. C*	Pressure Transducer for Mariner Venus/Mercury 1973, Flight Equipment, Detail Specification For

##### OTHER DOCUMENTS

Air Force

AFETRM 127-1	Range Safety Manual
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DOD

MIL Handbook 217B	Reliability Stress and Failure Rate Data for Electronic Equipment
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##### DRAWINGS

(TBD)*	Thruster Interface Control Drawing (TBD)
(TBD)*	PC Interface Control Drawing (TBD)

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\*JPL internal documents.

3.0 FUNCTIONAL DESCRIPTION

3.1 FUNCTION

The function of the electric thrust subsystem is to provide the total directed impulse ( $\sim 1.5 \times 10^7$  N-sec) required to accomplish a rendezvous with the comet P-Encke and, while operating, to provide control torques for vehicle three-axis attitude stabilization. To provide this impulse the electric thrust subsystem must convert solar panel derived electrical energy into the directed kinetic energy of ejected propellant, for a period of approximately 950 days. During this period, the available power for conversion to thrust will vary by about an order of magnitude.

3.1.1 SUBSYSTEM BLOCK DIAGRAM

The subsystem block diagram is given in Fig. 3.1-1.

3.2 REQUIREMENTS

The following design requirements for the electric thrust subsystem are a result of mission constraints, spacecraft physical and operational constraints, ground and in-flight environmental conditions, and the thrust subsystem characteristics:

- (a) The thrust subsystem shall be capable of delivering a total impulse of  $1.5 \times 10^7$  newton-seconds at a specific impulse of  $3000 \pm 100$  sec.
- (b) The thrust subsystem shall be capable of operating over a power range of 1.6 to 16 kw input power with an efficiency of not less than 63% at any operating point.
- (c) The thrust subsystem shall be capable of continuous operation over a period of 950 days without significant degradation in performance.
- (d) The thrust subsystem shall be capable of maintaining a vehicle attitude stable within a deadband of  $\pm 10$  milliradians while two or more thrusters are in operation.



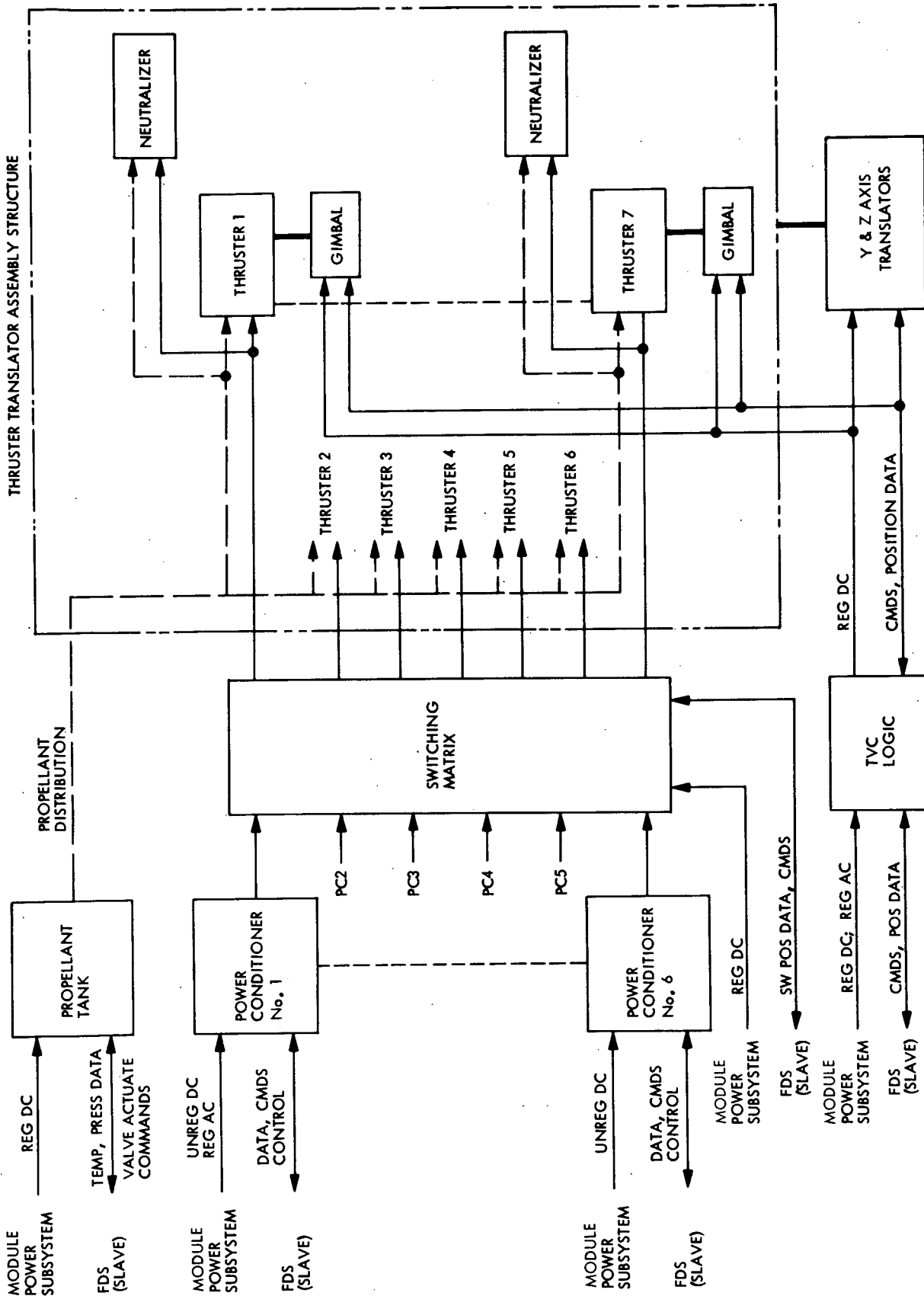


Figure 3.1-1. Thrust Subsystem Functional Block Diagram

- (e) The thrust subsystem shall be designed to dissipate its waste heat to the environment without requiring thermal coupling with other subsystems.
- (f) The overall reliability of the thrust subsystem shall be 0.96 for 950 days.
- (g) No single thruster or power conditioner failure shall cause a complete thrust subsystem failure.
- (h) Electromagnetic Compatibility - The entire thrust subsystem shall operate within the limits stated in (TBD).
- (i) The thrust subsystem shall be capable of adjusting the potential of the spacecraft relative to the ambient plasma potential.
- (j) Electronic equipment packaging shall meet the design requirements of TSS-220-1 including the packaging parts and materials specified in ZPP-2063-PMP. Where special requirements are recognized, calling for packaging techniques, different from those defined in TSS-220-1, the Electronic Packaging Engineer shall qualify and approve, in advance, the use of alternate techniques.

### 3.3 SUBSYSTEM ELEMENTS

#### 3.3.1 THRUSTER

##### 3.3.1.1 FUNCTION

The function of the thruster is to convert electrical energy supplied by the power conditioner into directed kinetic energy of the mercury propellant.

##### 3.3.1.2 DESIGN REQUIREMENTS

###### (a) Physical

(1) Thermal - The thruster shall be designed to operate in the array in a controllable manner while being exposed to the

vacuum, temperature, and solar environment associated with any heliocentric distance of 0.34 to 3.5 AU.

The thruster shall also be designed to warm up from a cold storage temperature of (TBD) °C to full operating temperature in not more than (TBD) minutes after turnon is commanded.

(2) Structural - The thruster shall be designed to pass the launch vibration and shock levels given in (TBD).

(b) Performance

(1) Stability - The thruster shall be designed to operate stably over at least a 2 to 1 throttling range. Oscillations of greater than 10 percent of the DC operating level shall not exist between 0 to 10,000 Hertz for stable operation.

(2) Optics - The thruster optics shall be designed to remain functional for at least 11,000 ±1000 hours of continuous operation at full operating power.

(3) Specific Impulse - The thruster shall have a true specific impulse of 3000 ±100 sec over the 2 to 1 throttling range.

(4) Efficiency - The thruster shall have an overall efficiency of not less than 71 percent at full operating power of 2630 watts input, and not less than 71 percent at an operating power of half full output power. The efficiency between full and half output power shall vary in a linear fashion. Over the operating life of the thruster, the efficiency shall not drop below 71 percent. Efficiency for the purposes of this specification is defined in paragraph 5.2.

(5) Reproducibility - Variations in thruster performance among "identical" thrusters under "identical" operating circumstances shall not exceed the efficiency and true specific impulse limits of operation.

(c) Mechanical

(1) Dimensions - The thruster shall have a maximum outside envelope diameter of 40 cm, and a maximum length of

(TBD) cm, exclusive of any neutralizer housings. Neutralizer housings shall have a maximum radius (from thruster axis) of (TBD) cm and a maximum intercepted angle of (TBD)°.

(2) Mountings - The thruster shall provide mounting shafts, or flanges to which such shafts can be mounted, at a location (TBD) cm from the rear of the ground screen, and at angular positions of (TBD)° and (TBD)° measured counter clockwise from the neutralizer centerline.

(3) Weight - The weight of the thruster shall not exceed (TBD) kg.

(4) Propellant Manifold Attachment - The thruster shall be designed to provide a single propellant manifold attachment. This shall be a M/F connector, size (TBD), part number (TBD), manufactured by (TBD).

(5) Emissivity - The thruster shall contain controlled emissivity surfaces.

(d) Electrical

(1) Isolation - The thruster design shall incorporate isolators which permit propellant storage and distribution at spacecraft ground, and biasing of the thruster body and the neutralizer relative to spacecraft ground. Leakage currents shall not exceed 20  $\mu$  amp.

(2) Connectors - The thruster shall be designed to accept input power through two M/F connectors, type (TBD) and (TBD), manufactured by (TBD).

(3) Electromagnetic Interference - Thruster EMI levels shall be consistent with overall subsystem limits shown in (TBD).

(e) Lifetime - The thruster shall have a design wearout lifetime of not less than 12,000 hours at full output power.

(f) Reliability - The thruster shall have a calculated random failure rate of not more than 5 per  $10^6$  hours. These calculations shall be based on lifetest data.

(g) Magnetic - The thruster residual magnetic field when not operating shall not exceed (TBD) nT at a location (TBD)

meters from the thruster in the plane of the thruster screen grid.

### 3.3.1.3 CALIBRATION REQUIREMENTS

(a) The thruster output (thrust and specific impulse) shall be calibrated over the full power range of the thruster. Calibration curves shall show the variation of true thrust and true specific impulse as functions of

- (1) Discharge current
- (2) Discharge voltage
- (3) Total accelerating voltage
- (4) Net accelerating voltage
- (5) Magnetic field strength
- (6) Ratio of flow rate through cathode to that of the main vaporizer
- (7) Neutralizer flow rate

at each point.

### 3.3.1.4 ACCEPTANCE CRITERIA

Acceptance test specifications shall be generated and an acceptance test performed on each thruster before integration into the subsystem.

### 3.3.2 POWER CONDITIONER

#### 3.3.2.1 FUNCTION

The function of the power conditioner is to convert the dc power supplied by the solar panels to the various ac and dc powers required to operate the thruster, and to provide control of thruster operation as specified below.

#### 3.3.2.2 FUNCTIONAL REQUIREMENTS

- (a) Electrical - The power conditioner shall convert the unregulated 200-400 V dc power supplied by the solar panels into

the multiple output, closely controlled voltages required by the thruster.

- (b) Thruster Output Control - The power conditioner shall contain all control loops necessary to regulate the thrust output of the thruster to a standard deviation of 2 percent and deviation of the time varying component of 3 percent with a correlation time of 5 days of an externally provided reference signal, and to regulate the specific impulse of the thruster to 3000 sec  $\pm$  100 sec for all required thrust levels.
- (c) Thruster Recycle Control - The power conditioner shall provide all circuitry required to suppress thruster arcing and to automatically restore thruster operation after arcing has been suppressed.
- (d) Thruster Grid Short Clearing - The power conditioner shall be capable of providing sufficient power to evaporate foreign metal particles of up to (TBD) sq cm in cross section area that may exist between the grids.
- (e) Self Protection - The power conditioner shall be self-protecting against any short circuits caused by thruster operation.
- (f) Telemetry - The power conditioner shall provide telemetry outputs sufficient to denote its status.

### 3.3.2.3

#### DESIGN REQUIREMENTS

- (a) Environmental - The PC shall be designed for operation in one atmosphere or in vacuum of  $10^{-6}$  torr or lower and in the temperature range of -20°C to 85°C. It shall not be damaged by exposure to -55°C, while in dormant state.
- (b) Part and Material Selection
  - (1) Electrical parts - Only electrical parts that have demonstrated high reliability shall be used. General guidelines of MIL Handbook 217 shall be adhered to.

(2) Packaging parts, materials and processes shall be selected from ZPP-2063-PMP.

(c) Electronic Packaging

(1) Consistent with special equipment requirements, power conditioner modules shall be packaged with electronic functional removable and replaceable modules and provide conductive heat paths to the TSS structure. Modules will serve as thermal conductors and be distributed to assist in the temperature control of all elements within the assembly. Adjacent modules shall make reciprocal use of radiative and conductive heat transfer to the maximum extent consistent with other factors. The modules shall provide a surface suitable for application of required temperature control finishes. Surface flatness and the number of fasteners used shall be compatible with thermal and temperature control design requirements.

(2) Provision for connectors where removable sections - modules - assembly - will be required for service or replacement.

(3) Provisions for cable routing and support shall be included in the PC structure design.

(4) The power transistor junction temperature shall not exceed 110° C operating at maximum with a 75° C shear panel temperature.

(5) Voltages in excess of 250v dc shall meet the design requirements of JPL Spec DM505139.

(6) The modules shall be mounted on a common frame of dimensions 0.51 x 1.40 m. designed to integrate directly with the s/c structure. The power conditioner structure shall be designed for a minimum resonant frequency of 200 Hz. The gains at resonant frequencies shall be less than 20 measured at discrete component parts relative to the s/c structure/PC interface with input vibration levels of 5 g's.

(7) The module shear panel temperature control surface shall carry in-plane shear loads of (TBD) and provide meteorite protection. A high conductivity aluminum shear plate with a thickness of 0.15 cm is recommended.

(8) The power conditioner structure shall provide a mounting surface of controlled flatness, roughness, and finish and interface attachment for a specified louver assembly to cover at least (TBD) percent of the surface area and designed to provide direct conduction path to the louver actuator.

(9) The radiating area for the power conditioner shear/temperature control surface shall be  $1000 \text{ in}^2$ .

(10) The power conditioner shall be installed as an assembly with only mechanical fasteners and connector mating required.

(11) The power conditioner assembly shall have controlled emissivity surfaces and be able to radiate from the two sides in plane with the shear/temperature control surface.

(d) Efficiency - The efficiency of the power conditioner, while operating at 100 percent beam power load, shall be not less than 92 percent. The efficiency at half load shall be not less than 92 percent. Over the operating life of the power conditioner, the efficiency shall not drop below these stated efficiencies. The efficiency shall not degrade when the input voltage varies between 200V and 400V.

Efficiency will be determined by measurement of the DC power by means of average reading DC-meters and of the AC power by means of true RMS meters.

(e) Weight - The weight of the power conditioner shall not exceed 13.6 kg.

(f) Reliability - The calculated reliability of the PC shall be not less than 0.96 for 10,000 hours operation at 75° C panel temperature.



- (g) Electromagnetic Compatibility - Power Conditioner EMI levels shall be consistent with overall subsystem limits shown in (TBD). Design of the PC shall be such that it will remain immune to EMI generated by neighboring PC/thruster sets.

The design shall consider and include EMI protection and suppression techniques including chassis ground interfaces with less than 25 milliohms dc resistance, cable routing and shielding, twisted wires, bundle routing and separation of signal, noisy, high and low voltage power cables, and shielded connector types.

- (h) Controls - Each power conditioner shall individually be controlled by means of digital commands, as required for turning the unit on and off. Analog beam and arc references shall be supplied externally, as 0-3V signals. (See also paragraph 4.3.3).

#### 3.3.2.4 ACCEPTANCE CRITERIA

Acceptance test specifications shall be generated and an acceptance test performed on each PC before integration into the subsystem.

#### 3.3.3 SWITCHING MATRIX

The switching matrix will be used to switch the electrical output of six (6) power conditioners to any of seven (7) ion thrusters. It will consist of six (6) rotary switches and a switching matrix assembly comprised of an integral chassis providing switch mounting, interconnections of switch outputs, and the interfaces to the power conditioners and thrusters.

#### 3.3.3.1 DESIGN REQUIREMENTS

In case of conflict with other published documents, not referenced herein, this specification is to take preference.

3.3.3.1.1 Environment

The switching matrix shall be designed for operation in a vacuum of  $10^{-6}$  torr and temperature range of  $-20^{\circ}\text{C}$  to  $85^{\circ}\text{C}$ ; it shall be capable of surviving temperatures from  $-55^{\circ}\text{C}$  to  $85^{\circ}\text{C}$ .

3.3.3.1.2 Switching Matrix Assembly Physical Data

Dimensions: 0.18 x 0.45 x 0.56 m

Weight: 15 kg

3.3.3.1.3 Reliability

The switching matrix shall be designed to operate for a period of three years with an actuation capability of at least 50 operations and a calculated reliability of 0.98.

3.3.3.1.4 Electromagnetic Compatibility

Switch EMI levels shall be consistent with overall subsystem limits shown in TBD.

3.3.3.1.5 Structural

The switching matrix must be capable of maintaining structure integrity under the launch environments specified in TBD.

3.3.3.2 FUNCTIONAL REQUIREMENTS

Switching shall be performed only when the power conditioners are turned off. Provisions shall be made to prevent connection of two thrusters to a single PC or two PCs to a single thruster.

Regulated DC voltage shall be supplied to the switching matrix from the power subsystem, as defined in paragraph 4.1.4.1.

The commands required to operate the switching matrix shall be as described in paragraph 4.3.4.

3.3.3.3 ROTARY SWITCH CONFIGURATION

Each switch consists of five (5) decks; each deck has four (4) input contacts. Seventeen input contacts will be used for

processing the power from each of the power conditioners; one contact shall be utilized as the position feedback to data system (see paragraph 4.3.4); two contacts shall be provided as spares. Two decks shall handle lines at high voltage, three decks shall take care of the lines that operate near S/C ground potential. All contacts will be rated at 5A, except these assigned to the arc supply, which will be rated at 22A.

#### 3.3.3.4 ACCEPTANCE CRITERIA

Acceptance test specifications shall be generated and an acceptance test performed before integration into the subsystem.

#### 3.3.4 THRUST VECTOR CONTROL

##### 3.3.4.1 FUNCTION

The function of the thrust vector control (TVC) element is to provide control torques about the three primary s/c axes utilizing the thrust of the electric thrust subsystem. This will be accomplished by translating the average thrust vector of all operating thrusters parallel to two axes through the s/c center of mass, and by gimbaling individual thrusters to provide a couple about the third axis.

The TVC will operate in closed loop with the vehicle attitude control star tracker and sun sensors. Vehicle attitude as sensed by the star tracker and sun sensors and vehicle rate data by the rate estimators will be used to command repositioning of the thruster translator assembly structure and gimbaling of thruster pairs. Proportional control will be used, with stepper motor rates for control about any axis proportional to the error in that axis.

##### 3.3.4.2 DESCRIPTION

The TVC consists of the following components.

(1) Gimbal actuators, which mount to the Thruster Translator Assembly (TTA) structure and rotate the individual thrusters.

(2) Translator carriage, containing the bearings through which the translator rods move.

(3) The translator rods, which connect the TTA structure to the translator carriage, and the translator carriage to the SEP module primary structure.

(4) The translator actuators, which move the TTA along one pair of translator rods through the carriage, and move the carriage along the other set of translator rods.

(5) The TVC logic, which controls and directs the operation of the actuators.

### 3.3.4.3

#### REQUIREMENTS

##### (a) Performance

(1) The TVC shall be capable of maintaining the space vehicle fixed within a deadband of no more than  $\pm 10$  milliradians about each space vehicle axis, in limit cycle operation when two or more thrusters are in use.

(2) The TVC shall be capable of nulling out a disturbing impulse of as much as 5.12 newton-meter sec without violating a deadband of  $\pm 10$  milliradians.

(3) The TVC shall have an operating lifetime of at least  $1.0 \times 10^8$  actuations per axis.

(4) The TVC shall have maximum peak power demands of 68 watts at regulated DC voltage and 32 watts delivered at a voltage frequency of 2.4 kHz.

##### (b) Structural

All components will be designed to maintain structural integrity under the launch environment specified in TBD.

(c) Thermal

(1) The gimbal actuators shall be designed to operate in a thermal environment as specified in TBD.

(2) The translator actuators shall be designed to operate in a thermal environment as specified in TBD.

(3) The TVC logic shall be designed to operate in a thermal environment as specified in TBD.

(d) Electrical

(1) The translator and gimbal actuators shall operate on a 28V regulated DC voltage input.

(2) The TVC logic shall operate on a 50V, 2.4 KHz square wave AC voltage input.

3.3.4.4

TVC OPERATION

The TVC will be operational at all times the thrusters are on during the mission, with the exceptions that if the deadband of  $\pm 10$  milliradians is exceeded, the reaction control system (RCS) will be automatically activated. Activation of the RCS will automatically suspend TVC closed loop operation and command the TVC to return to zero position. When position and rate fall within their RCS deadbands, TVC operation will automatically resume.

3.3.4.5

ACCEPTANCE CRITERIA

Acceptance test specifications shall be generated and an acceptance test performed before integration into the subsystem.

3.3.5

PROPELLANT TANKAGE AND DISTRIBUTION

3.3.5.1

FUNCTION

The propellant tankage and distribution element contains the thrusters' propellant and pressurant storage reservoirs and delivers mercury propellant to the thrusters at positive pressure.

Pressurization is accomplished by vapor of Freon TF, whose vapor and liquid states are in equilibrium over a temperature range compatible with external environmental temperatures. Fig. 3.2.5-1 indicates the interfaces of the subsystem with the spacecraft. The principal components of this element are the (1) tank with an expulsion bladder (diaphragm), (2) pressure and temperature transducers, (3) tank valve, (4) propellant distribution network (including, as required, feedlines, flow distributors, manifolds, and propellant shutoff valves), (5) pressure control (tank) heaters (and any required insulation), and (6) thermal control components as may be required for maintaining the mercury within the tank and distribution network in a liquid state throughout the required portion of the mission. The propellant tankage and distribution network consists of these components along with the structure and cabling to integrate them into the SEP module. The element itself is not considered capable of being easily removed intact from the SEP module since the tank assembly is necessarily required to be physically separated from the thruster array.

The tank will have the capability of being off-loaded after being integrated into the propulsion system, the space vehicle and mated to the launch vehicle.

#### 3.3.5.2 DETAILED ELEMENT DESIGN DESCRIPTION

The components as described below are intended to meet the requirements and constraints discussed below (including compatibility with all fluids with which they come in contact), and all other requirements resulting from physical and operating constraints described in TBD, launch vehicle characteristics described in TBD, and the ground, launch, and in-flight environment specified in TBD.

##### 3.3.5.2.1 Tank

The tank will be designed to contain the propellant and pressurant at a maximum allowable pressure and temperature of

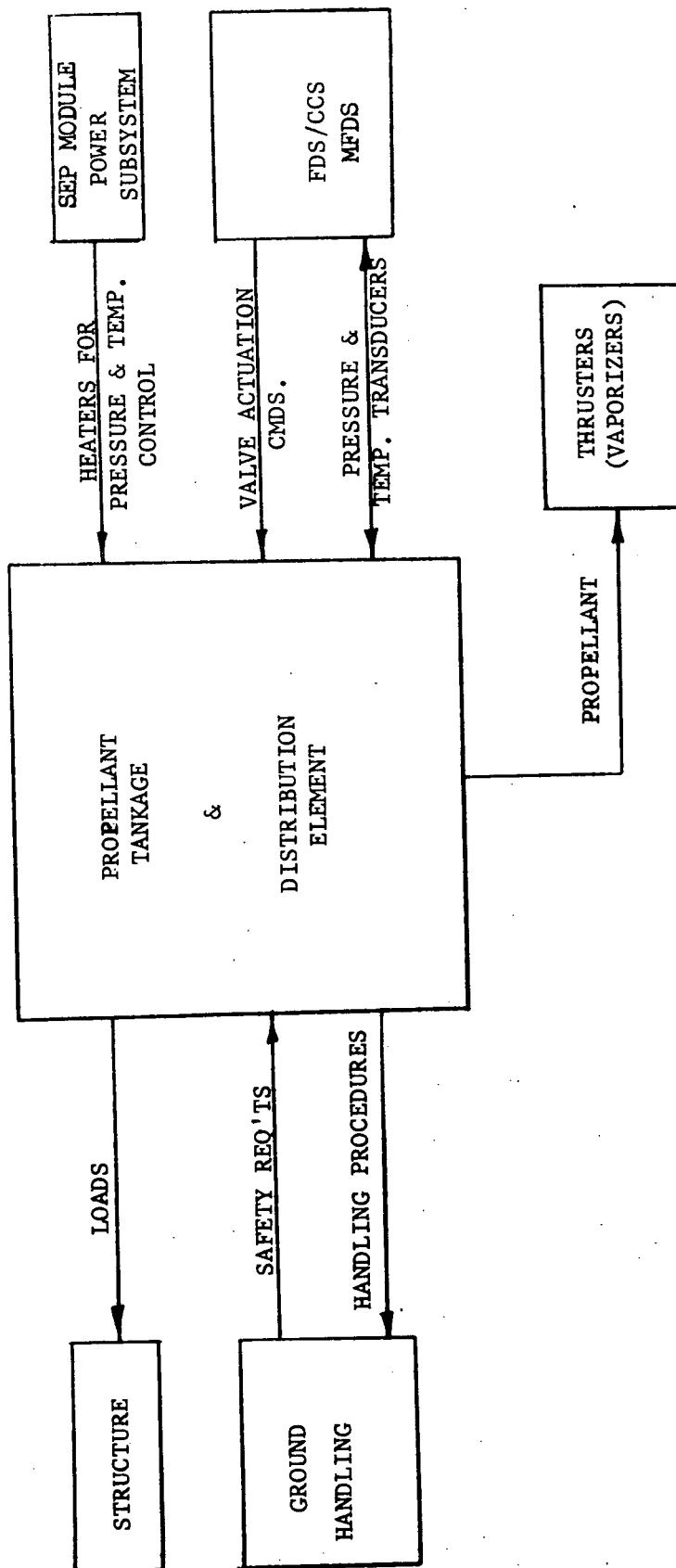


Figure 3.2.5-1. Propellant Tankage and Distribution Interfaces

$2.76 \times 10^5 \text{ N/m}^2$  at  $82^\circ\text{C}$  (40 psia at  $180^\circ\text{F}$ ). Approximate operating range to insure all mercury is expelled will be  $3.45 \times 10^4 \text{ N/m}^2$  at  $18^\circ\text{C}$  (5 psia at  $65^\circ\text{F}$ ) to  $2.07 \times 10^5 \text{ N/m}^2$  at  $68^\circ$  (30 psia at  $155^\circ\text{F}$ ).

The tank will consist of two hemispherical shells fabricated from Type 304 stainless steel or other material which can be properly and satisfactorily shown to be compatible with mercury and Freon TF at the required pressures and temperatures. Internal to the tank is an elastomer diaphragm which isolates the pressurant from the propellant and forces the propellant out of the tank through the distribution network to the thrusters' vaporizers, and by continuously being acted on by the Freon vapor pressure, continues to drive mercury from the tank as the propellant is consumed. The diaphragm will be fabricated from a high-strength, sulfur-free neoprene, or other suitable equivalent which is compatible with mercury and Freon TF at the operating temperatures and pressures. Expulsion efficiency of the tank will be TBD.

#### 3.3.5.2.2 Pressure and Temperature Transducers

Freon TF vapor pressure will be monitored from pressure data signals from a pressure transducer. Operation of the tank heater (see paragraph 3.2.5.2.6) to maintain the pressure within the required range is based on these signals. Operational range of the transducer will be  $0\text{-}3.45 \times 10^5 \text{ N/m}^2$  (0-50 psia). The transducer design will be similar to that specified for use in Mariner 1973 (see Spec. #CS 506216 Rev. C) except that it will meet the specific requirements imposed herein. Temperature transducers (thermistor type) will be used to monitor tank temperature and also the feedlines as deemed necessary.

#### 3.3.5.2.3 Tank Valve

A valve is required at the tank propellant-outlet to prevent propellant exiting the tank at launch, and allow propellant to exit



from the tank to the thrusters as required during operation. It will be a latching solenoid type valve requiring power only during OPEN or CLOSE operations. Actuation will be commanded by signals from the FDS slave (see paragraph 4.3.6).

#### 3.3.5.2.4 Propellant Distribution Network

The propellant distribution network contains necessary feed tubes, manifolds, valves, fittings, and flow distributors needed to properly distribute the propellant exiting from the tank to the thrusters' vaporizers. The feed tubes across the translating interfaces will be a coiled design. Each of the feed tubes across the gimbal interfaces will be coiled design and will exit from fittings at the thruster array structure in a direction perpendicular to the thruster gimbal axis to minimize torsional stresses during gimbal operation.

All components and tubing will be fabricated from Type 304 stainless steel. The coiled tubing across the movable interfaces will be fully hardened. All tubing will be properly supported to the structures. Welded or brazed tubing and fittings are to be used whenever possible. Metal seals are to be employed where welded and brazing are impractical in order to minimize the effects of irradiation, hard vacuum, temperature, and long term storage. Shutoff valves as may be needed for shutting off propellant flow to thrusters with malfunctioned vaporizers shall be either solenoid-latching type or normally-open explosive type.

#### 3.3.5.2.5 Pressure Control (Tank) Heater

Heater elements will be required at the tank to maintain Freon vapor pressure at valves specified in 3.3.5.2.1. Total power required for pressure control will be TBD watts.

3.3.5.2.6 Thermal Control Heaters

Heater elements will be required at TBD to insure that local condensation of mercury does not occur throughout the thruster operation. Total power requirements is TBD watts.

3.3.5.3 SUBSYSTEM OPERATIONS

The following paragraphs describe the operational requirements of the propellant tankage and distribution system during the mission to include pre-launch.

3.3.5.3.1 Pre-Launch

Procedures for assembly and installation of the subsystem into the spacecraft, including checkout, prior to launch are described in TBD. In its assembled position in the spacecraft while on the launch stand and until initiation of the launch sequence procedures the subsystem will be maintained at TBD. The tank valve will be in CLOSE position. Any other valves will be in OPEN position. If at any time before launch mission requirements dictate a reduction in propellant loading, off-loading will be performed at the launch stand per procedures described in TBD.

3.3.5.3.2 Launch

Following launch and prior to the initiation of thruster startup procedures the subsystem temperature will not be allowed to fall below  $-39^{\circ}\text{C}$ .

3.3.5.3.3 Startup

Propellant flow as required in the startup sequence will be initiated by commanding the tank valve OPEN.

3.3.5.3.4 Normal Operation

Operation during the normal mode consists of maintaining the subsystem temperatures at the nominal operating values. If at any time during the normal thruster operation a serious vaporizer

leak should occur, propellant flow to the affected thruster will be terminated by commanding the appropriate propellant valve CLOSE. At no other time during the programmed thrust sequence will the propellant valves be closed except for such malfunctions.

#### 3.3.5.3.5 Thruster Shutdown

At the completion of the programmed thrust sequence the tank valve will be commanded CLOSE and all power to heaters removed.

#### 3.3.5.5 ACCEPTANCE CRITERIA

Acceptance test specifications shall be generated and an acceptance test performed before integration into the subsystem.

#### 3.3.6 THRUST SUBSYSTEM CABLING

##### 3.3.6.1 GENERAL

##### 3.3.6.1.1 Intrasystem Cabling

In conjunction with other cabling, the flight electrical cabling provides electrical interconnections for those interfaces identified in paragraph 4.6.

##### 3.3.6.1.2 Thrust Subsystem Cabling

All cabling required to make electrical connections to the assemblies and independently mounted units is included in the subsystem cabling. Not included are wires within subassemblies (such as internal to a Power Conditioner module). Cabling fabricated to designs based on detail electrical requirements provided on subsystem input diagrams by the subsystem cognizant engineers.

##### 3.3.6.1.3 Subsystem Level Circuits

Only those subsystem level flight electrical interface circuits identified as part of the Functional Diagram and Interface Lists

and those direct access circuits identified as part of the Support Equipment Functional Block Diagrams and Interface Lists will be mechanized in the flight electrical cabling. The circuit name listed for each circuit shall be the approved circuit identifier and is the only name by which the circuit is to be referenced on cabling documentation.

#### 3.3.6.1.4 Cabling Documentation

A master cabling interconnect diagram shall be prepared showing each cable and cable harness in relation to the associated electronic assemblies, sub-assemblies and interface connectors, utilizing drawing numbers, reference designations and other appropriate means. All harnesses shall be documented by a standard format wiring harness diagram. Cabling installation drawings shall be provided for each harness requiring support from the primary or secondary Thrust Subsystem structure. The interface circuits identified in the functional block diagrams shall be further described and defined in Circuit Data Sheets. The Circuit Data Sheets shall specify twisting, shielding, cable bundling and certain other information for each interface circuit. All cables and connectors shall be identified in a permanent and legible manner in accordance with approved marking processes.

#### 3.3.6.2 PERFORMANCE PARAMETERS

##### 3.3.6.2.1 Performance Margins

Performance margins and component derating and conformance with the requirements for the spacecraft are specified in TSS-2009-1, Thrust Subsystem Cabling, Cable and Harness Assemblies, Design Requirement. Derating applying to wires and connectors shall allow particularly for the effects of the thermal and vacuum environments, such as heating by the Sun, cooling in a shadow and wire bundles flexing during flight. Redundant cabling conductors may be employed where necessary to achieve acceptably small voltage drops in the wiring. The

quantity and wire gage of the redundant conductors shall be consistent with the cable resistances (which are to be allocated to achieve a total loop resistance not to exceed that specified in the applicable Circuit Data Sheets). Allocation of cable resistance shall be an integral part of the cabling design process. Redundant cabling conductors introduced to improve reliability must satisfy the requirements of paragraph 5.3. Electrical cross-coupling of cabling circuits will be controlled within the limits required for reliable system operation by providing isolation of incompatible circuits, using physical separation, twisting and shielding.

3.3.6.2.2 Quiet Signal Circuits

Twisting and shielding shall be as specified in the Circuit Data Sheets. Physical separation of quiet circuits from other circuits shall be provided. Quiet circuits shall be identified in the circuit data sheets.

3.3.6.2.3 Noisy Signal Circuits

Bundling of these circuits will be provided to maintain separation from other circuits. Noisy circuits shall be identified in the circuit data sheets.

3.3.6.2.4 Power Circuits

Power will be distributed to the various elements of the Thrust Subsystem during cruise as required. Power cables shall be designed to minimize electrostatic and electromagnetic coupling with other circuits. Power losses in the conductors shall be minimized. Connectors for power circuits interfaces shall have current ratings, insulation and other characteristics compatible with the voltage and currents being handled. In particular, the power cables shall be grouped into Low Voltage and High Voltage cables, and shall be bundled and routed separately. Separate connectors shall be employed for Low and High Voltage circuits wherever possible.

3.3.6.2.5 Pyrotechnic Circuits

Cabling utilized for electroexplosive device firing circuits shall exclude undesired electrical energy by the use of complete and continuous shielding from the pyrotechnic control unit to the device housing, including exclusive use of approved type of connectors.

3.3.6.2.6 High Voltage Protection

Circuits utilizing instantaneous voltage differentials in excess of 250V peak between two or more conductors shall conform to DM 505139, High Voltage Protection. At AC frequencies higher than 60 Hz, the voltage levels of concern shall decrease in accordance with DM 505139.

3.3.6.2.7 EMI Protection

Electromagnetic interference shall be controlled by appropriate design practices which shall include physical and electrical separation of various circuits, shielding of signal lines, elimination of ground loops by proper selection of grounding locations, use of twisted cable configurations for AC lines, and termination of sensitive circuits in separate connectors. The use of filter pin connectors may be allowed when maximum voltage drop and power loss requirements are met. The cabling EMI design practices shall be compatible with the overall electrical design requirements of paragraph 4.3.9 (TBD).

3.3.6.2.8 Acceptance Criteria

Acceptance test specifications shall be generated and an acceptance test performed before integration into the thrust subsystem.

3.3.6.3 CABLING CHARACTERISTICS AND RESTRAINTS

3.3.6.3.1 General

Cable harnesses shall function without degradation of performance or reliability during exposure to the applicable environments

encountered during the mission. The cable harnesses shall be designed to withstand the environmental stress encountered on Earth during testing, checkout and launch as well as those environments encountered in space. The effects of the space environment upon all components and materials used in the cabling shall be evaluated. The following electrical, mechanical and thermal restraints shall apply to the design and fabrication of the Thrust Subsystem Cabling.

3.3.6.3.1.1 Electrical

The cable harnesses shall conduct the power and signals in a manner compatible with the requirements of the source unit and the destination unit. Prime considerations shall include:

- (a) Control of magnetic and electrostatic interference.
- (b) Voltage drop in conductors.
- (c) Voltage drop in grounding connections.
- (d) Insulation electrical characteristics.
- (e) Reduction of subsystem power losses.
- (f) Thermal control.

3.3.6.3.1.2 Mechanical

Mechanical aspects of cable harnesses include insulation strength (tensile and bearing), tolerance to vibration, adequate support, accessibility during construction, test and rework. In the physical installation, an attempt shall be made to:

- (a) Avoid areas where adverse conditions may exist, or if unavoidable, adequate protection shall be provided.
- (b) Avoid interference with optical instruments. Test connector location shall provide for ease of access and preclude damage to any instruments when test harnesses are connected.

## 3.3.6.3.1.3 Thermal

Harness design, especially harnesses which must flex during flight, shall consider thermal cycling effects caused by alternate Sun and shade exposure. Appropriate temperature control shall be provided. The current carrying capacity of wires and connectors shall be derated to meet anticipated combinations of thermal and vacuum conditions of space, launch, and injection. De-rated current values are presented in Table I.

Table I. Current Carrying Ratings (Continuous Duty) of Wires and Cables (TFE Jacket)

Wire Size (AWG)	Continuous-Duty Current (amperes)	
	Single-Wire In Free Air	Wires & Cables Conduit or Bundles
Silver-Coated Copper	12	41.0
	14	32.0
	16	22.0
	18	16.0
	20	11.0
	22	7.0
Copper Alloy	24	2.9
	26	1.9
	28	1.2
	30	0.7
	32	0.44

These current values are specified for a temperature of 200°C (392°F) maximum allowable conductor temperature and maximum ambient temperature around wires of 155°C (311°F).

Current values are for each wire in the conduit or bundle. Copper alloy conductivity = 0.84 x copper conductivity.



3.3.6.3.2 Weight

The design shall emphasize the application of components and techniques that result in the minimum cabling weight compatible with the subsystem and reliability requirements. The routing of wire runs shall be selected as the shortest possible, compatible with the circuit separation requirements of paragraph 3.3.6.2. The total weight of the cabling subsystem shall not exceed the allocated weight specified in paragraph 6.1.

3.3.6.3.3 Power Loss

Power losses in the subsystem cabling shall be minimized by appropriate selection of conductor gage compatible with current handling limitations of paragraph 3.3.6.3.1.3. Power losses shall be traded off against cabling weight.

3.3.6.3.4 Voltage Drop

The cabling design shall satisfy the voltage drop requirements by compliance with the circuit resistance limit specified in the applicable circuit data sheets, except that in no case shall the voltage drop in any circuit exceed 2%.

3.3.6.3.5 Clearance

Cabling shall be routed to provide adequate clearance required for articulating mechanisms, antennas, sensor fields of view, attitude control gas jet exhaust, and thruster beam.

3.3.6.3.6 Articulation

Wiring across articulated interfaces shall be given special design consideration to insure reliable operation over the required range of motion under all anticipated environments.

3.3.6.3.7 Cable Jacketing

Cables shall be protectively jacketed in regions of possible abrasion or stress concentration.

3.3.6.3.8 Connector Interchange

Every effort shall be made to select connectors sufficiently dissimilar to prevent incorrect coupling to connectors in the vicinity.

3.3.6.3.9 Connector Gender

In general, the pins or sockets of a connector shall be selected so that the power flow is from socket to pin side of connector. Connector contacts on direct access connectors (unmated during flight) shall be nonprotruding socket types in hardmounted receptacles. When in-flight separation connectors are employed the power flow shall control the selection although in general the pin type contacts can be assigned to the TSS side of the interface.

3.3.6.3.10 Connector Joint Usage

In general, no more than two elements or assemblies of the thrust subsystem shall be interconnected through any given connector (i. e., each power conditioner shall have separate connector inputs to the switching matrix assembly).

3.3.6.3.11 Connector Mounting

In general, connector shells shall be conductively mounted to the associated mechanical element (subchassis, bracket, etc.).

3.3.6.3.12 Test Receptacles

Direct assess test circuits between the flight thrust subsystem and the support equipment shall be carried by the subsystem cabling to test receptacles mounted directly on the electronic assemblies as required, consistent with thrust subsystem design criteria.

3.3.6.3.13 Cable Interchangeability

All cable and harness assemblies having the same part number shall be directly and completely interchangeable with respect to installation and function.

## 3.3.6.3.14 Splices

Splicing (the joining together of two or more wires to form an electrical junction at a point in the cabling) shall be kept to a minimum. Only butt splices shall be used and their use shall be limited to multiple interconnections (i. e. , chassis ground) where a significant saving in weight or quantity of wire and connectors can be realized, or to interconnection of components provided with pigtailed leads (i. e. , transducers). Circuit integrity and maintainability shall not be impaired by the use of splices. Splices shall be located in rigid or semiflexible portions of the cabling.

## 3.3.6.3.15 Connector Contact Assignment

The following criteria shall govern connector contact assignment.

- (a) Spare or Unused Pins - Connectors shall have the facility of accepting additional circuitry at a later date. Spare pins may be used to separate incompatible groups of wiring. If reserved for future assignment only, the pin should be in an accessible insert location. Normally, the keyed side of the circular connector is up during fabrication; therefore, pins on the upper and outer periphery shall be left unused for future applications.
- (b) Adjacent Pins - Where possible, adjacent pins shall be used for bussed, spliced, twisted and shielded functions.
- (c) Function Grouping - Within a given connector, various circuit functions can exist. These functions are defined by type and level of signal or power and categorized into groups. Functions within a group shall be assigned adjacent pins and isolated from incompatible groups by use of spare pins. The EMI hazard shall be considered when selecting contact assignments.
- (d) Socket/Pin Selection - Where feasible, the pins or sockets of a connector shall be selected so that the power flow is

from socket to pin side of the connector. Connector contacts on the thrust subsystem half of direct access connectors (unmated during flight) shall be nonprotruding socket types for safety reasons.

- (e) Conductors Per Contact - Not more than one conductor shall terminate at any single connector contact.

3.3.6.3.16 Terminal Boards

Terminal boards shall not be used for interconnection of interface functions.

3.3.6.3.17 Connector Types

Removable crimp contact type connectors shall be employed for the thrust subsystem cabling. Special high voltage connectors shall be provided for those circuits identified in paragraph 3.3.6.2.6. Solder type connectors may be used for interconnection of wires on the module or subassembly side of the interface to an assembly harness if the connector is compatible with the subsystem connector.

3.3.6.3.18 Materials, Parts, and Processes

Unless otherwise specified on the applicable wiring harness diagram or cable installation drawing, all parts, materials, and processes used on the thrust subsystem cabling shall be in conformance with JPL Specification ZPP-2063-PMP.

3.3.6.4 DESCRIPTION OF CABLES

3.3.6.4.1 Block Diagram

A simplified block diagram of the thrust subsystem cabling appears in Figure 3.3.6-1.

3.3.6.4.2 Power Conditioner Compartment Ring Harness

The RING (Remote Interconnecting Grouping) harness shall be mounted on a support structure located within the power

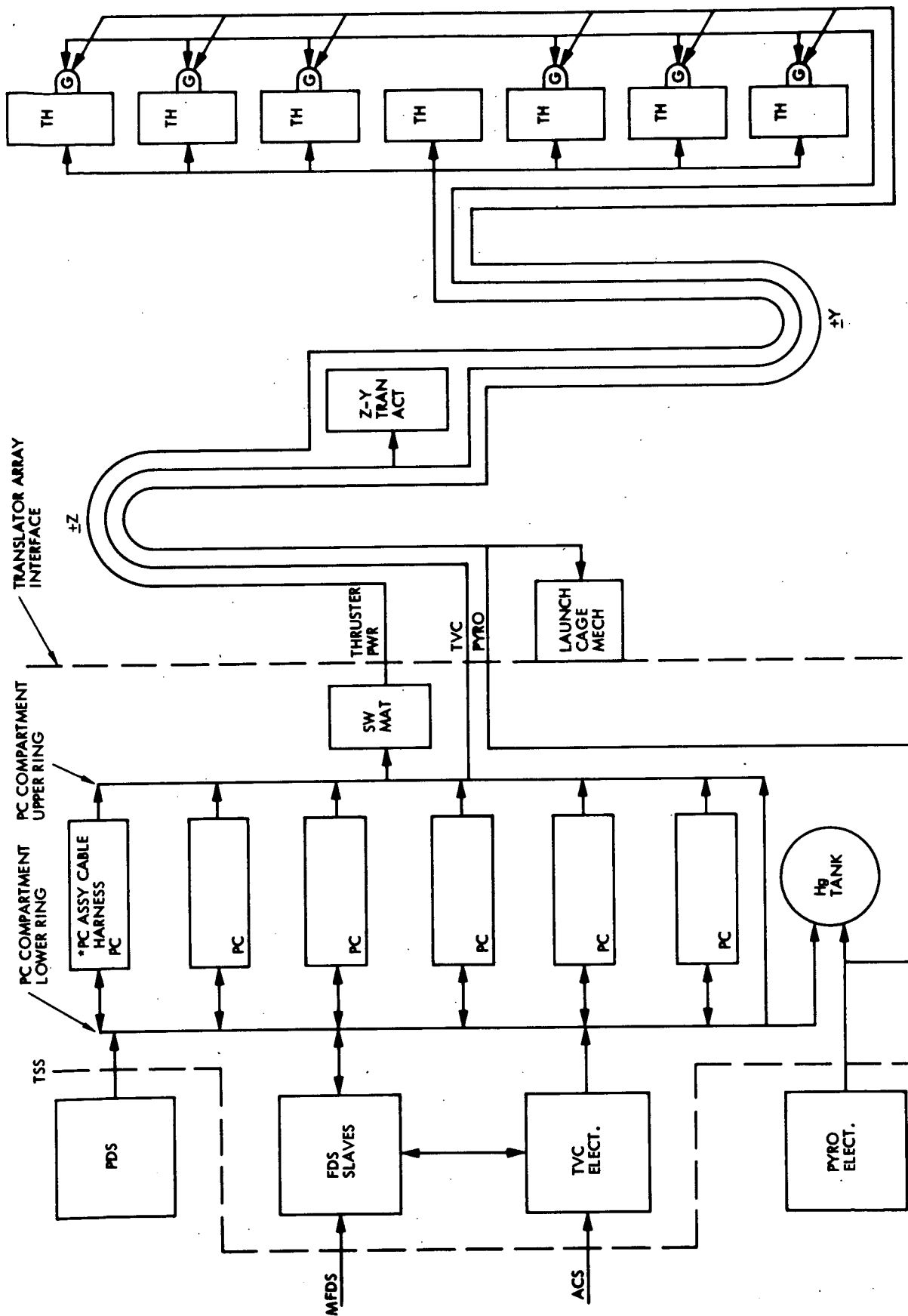


Figure 3.3.6-1. TSS Simplified Cabling Block Diagram

conditioner compartment primary structure. This harness shall interconnect the power conditioners to the switching matrix assembly and the power distribution subsystem of the attachable module. To control cross coupling, the design of the RING harness shall provide for physical separation of wire groups, and wherever possible, the associated connectors for noisy-signal, quiet-signal, low voltage-power, and high voltage-power circuits. The connectors of the RING harness shall either be hard-mounted to the supporting structure or connect to hard-mounted connectors on the power conditioners. The connectors shall be accessible from the sides of the power conditioner compartment (with the thermal shield removed). This harness shall also incorporate connectors and cables interfacing with the mercury propellant tank which is mounted in the compartment. Additional telemetry, control, and power circuits from the attachable module electronics shall be routed through this harness to the power conditioners, switching matrix assembly and thruster array.

3.3.6.4.3 Power Conditioner Assembly Harnesses

The power conditioner assembly harnesses shall be designed to interconnect integral shear plate or plug-in modules. The harnesses shall interconnect the power conditioner to the power conditioner compartment RING (Remote Interconnecting Grouping) harness and interconnect the individual modules and shall present minimal interference with the installation and removal of the modules. The harness shall be designed to be installed on the individual power conditioners prior to integrating the power conditioner assembly with the thrust subsystem structure. Direct access test receptacles shall be included in the harness and shall be hard mounted on the assembly structure. A total of six (6) of these harnesses is required.

3.3.6.4.4 Thruster Array Harness

The Thruster Array Harness provides the interconnection of electronic equipment mounted on the Thruster Array Structure.

This includes the seven (7) thrusters, six (6) gimbal actuators, two (2) translator actuators, and any remote sensors or transducers which may be required. The cabling shall be mechanized by the use of a ribbon cable configuration to provide the transfer of circuits across the translation X-Y plane. These cables shall be supported in opposing cable troughs mounted on the thruster translator assembly structure which permit a rolling action of the cable during orthogonal translation. The cables shall be supported and mounted by clamps and bracketry directly to the array structure and routed in the ribbon (flat) condition to within 25 cm (9.84 in.) of the electronic equipment interface. Assignment of circuits in the ribbon cable shall consider routing and breakout sequence, installation and access, and circuit separation for EMI and high voltage protection.

#### 3.3.6.4.5 Pyrotechnic Harnesses

Harnesses for actuation of electroexplosive devices and valve solenoids shall be supplied as separate cabling which does not serve any other function. These cables shall be physically separated but routed and installed similarly to the RING and thruster array harnesses. These cables will serve such functions as unlatching of the translator caging device, disruption of the mercury feed to a malfunctioning thruster, or initiation of in-flight separation or deployment devices as required. Pyrotechnic cables shall conform to the applicable requirements of the Air Force Eastern Test Range Safety Manual AFETRM 127-1.

#### 3.3.7 TSS STRUCTURE

##### 3.3.7.1 FUNCTION

The TSS structure structurally integrates the TSS components and elements into the SEP module and in some cases serves as components for thermal control and SEP module structure. The structure must meet environmental constraints specified in TBD.

The TSS structure is comprised of the following components:

1. Thruster Structure
2. Power Conditioner Structure
3. Thruster Translator Assembly Structure
4. Propellant Tank Support Structure

3.3.7.1.2 Thruster Structure

The thruster structure structurally integrates each thruster's components into the configuration. It shall provide mounting shafts or flanges to which shafts can be mounted, in order that proper interfacing with the TVC gimbal actuators can be accomplished.

3.3.7.1.3 Power Conditioner Structure

The PC structure is primarily in the form of shear plates and integrates the PC components into modules as required. These plates also serve as; 1) shear plates for the SEP module structure and 2) mounting surfaces for mounting of thermal control components.

3.3.7.1.4 Thruster Translator Assembly Structure

The Thruster Translator Assembly (TTA) is that assembly which includes the thrusters, TVC components (less TVC logic), cable troughs for carrying propellant feedlines and cabling across the articulating interfaces, propellant feedlines and cables, any thermal control items and the TTA structure. The function of the TTA structure is to integrate these components and elements into the assembly, and also the assembly onto the SEP module. It shall include latches that secure the TTA to the SEP module structure in such a way lateral and axial motion is prevented during this event.



3.3.7.1.5 Propellant Tank Support Structure

The function of the propellant tank support structure is to integrate the propellant tank into the SEP module.

3.3.7.2 REQUIREMENTS

TBD (Strength, stiffness, mounting requirements, etc.)

4.0 INTERFACES

4.1 SEP MODULE POWER SUBSYSTEM INTERFACE

The SEP module power subsystem shall provide power for the operation and maintenance of the thrust subsystem during Sun orientation periods.

The power provided shall have the following voltage waveforms:

- (a) Unregulated DC,
- (b) Regulated DC, and
- (c) Regulated AC at 2.4 kHz square wave.

The unregulated DC voltage for the thrust subsystem power conditioners shall be supplied by the solar arrays. Power at the regulated voltages shall be supplied by the power conditioning units in the SEP module power subsystem.

4.1.1 SOLAR ARRAY BUS

The power output of the solar arrays at 1 AU and the beginning of the mission shall be 20 kW. The 20 kW output shall include an 18% contingency for space degradation. The solar arrays consist of series parallel combinations of N on P solar cells. The output impedance of the source will vary depending on the power requirements (operating point on the EI characteristic of the source).

The solar arrays are fixed polarity DC power sources which are current limited. The power conditioner shall not feed back current of reversed polarity to the input power source. Any network of filtering necessary to assure compatibility with the solar array power source shall be part of the power conditioner. The peak current ripple on the solar array line shall be not greater than 5% of the average input current.

In order to avoid transient overloading of the solar panels, the PC peak power demand shall not exceed the steady state demand by more than 1%.

#### 4.1.2 UNREGULATED POWER BUS

Power at an unregulated voltage shall be supplied to the thrust subsystem for the operation of the propulsion power conditioners. The unregulated bus voltage shall be between 200 to 400 volts during Sun tracking periods. The maximum voltage of 400 volts shall be limited through the use of active shunt regulators.

##### 4.1.2.1 POWER CONDITIONER INTERFACE

The primary voltage to the propulsion power conditioners shall be unregulated between 200 to 400 volts. The power requirements of each power conditioner shall be as follows:

Peak power - TBD watts

Operating power - 2860 watts

The power conditioner shall provide stable operation over the variable output impedance of the solar array. The expected transient response of the solar array is between 1 to 60  $\mu$ s.

The power required by the power conditioner at voltages below 200 volts shall be rapidly diminished to zero. The exact voltage at which the power conditioner power requirements is zero shall be established in the design, but shall not be below 150 volts. The input current ripple (peak to peak) shall be limited to 1% of the operating current under all conditions, including transient.

#### 4.1.3 2.4 KHZ AUXILIARY POWER BUS

Power with a 2.4 kHz, 50-volt RMS, square wave voltage waveform shall be provided as auxiliary power for the subsystem. The 2.4 kHz voltage shall be available after Sun acquisition and during Sun tracking periods, and shall have the following output characteristics:

Voltage: 50-volt RMS, square wave

Voltage tolerance: +3%, -4%

Rise and fall time:  $5 \pm 4 \mu s$   
Spikes: within the limits of the EMI spec  
Frequency: synchronized -  $2.4 \text{ kHz} \pm 0.01\%$   
free running -  $2.4 \text{ kHz} \pm 6\%$

4.1.3.1 POWER CONDITIONER INTERFACE

The propulsion power conditioner may require 2.4 kHz regulated square wave voltage prior to turn-on and during operation. The power required by each power conditioner during operation shall be limited to 2.5 watts maximum. The power requirements shall be reduced to zero when the power conditioner is off.

4.1.3.2 THRUST VECTOR CONTROL INTERFACE

The thrust vector control unit shall receive 2.4 kHz regulated square wave voltage for the operation of the electronics. The power requirement shall be 32 watts.

4.1.4 REGULATED DC AUXILIARY POWER

Auxiliary power delivered at a regulated voltage of 28 volts DC shall be available for use by the propulsion supporting subsystems and for heating of the propulsion power conditioners which are not in operation. Regulated DC voltage shall be available after Sun acquisition and during Sun tracking periods. The ripple and spikes present on the bus shall be as follows:

Peak-to-peak ripple: 1% of average value  
Spikes: within the limits of the EMI spec

4.1.4.1 SWITCHING MATRIX INTERFACE

Regulated DC voltage shall be supplied to the switch actuators as commanded by the FDS.

4.1.4.2 THRUST VECTOR CONTROL INTERFACE

Regulated DC voltage shall be supplied to the driving motors of the thrust vector control unit as required.

4.1.4.3 POWER CONDITIONER THERMAL CONTROL INTERFACE

Power at a regulated DC voltage shall be supplied to heaters for thermal control to maintain the minimum temperature of each PC unit.

4.1.4.4 PROPELLANT TANKAGE AND DISTRIBUTION INTERFACE

Power at a regulated DC voltage shall be supplied to the propellant tankage and distribution subsystem for operating the valves and for the temperature control.

4.2 SEP MODULE STRUCTURE INTERFACE

4.2.1 THRUSTER TRANSLATOR ASSEMBLY STRUCTURAL INTERFACE

The thruster translator assembly (TTA) has a structural interface with the SEP module primary structure. The TTA interface is controlled by the following factors:

4.2.1.1 MOUNTING CONFIGURATION: The TTA will be attached to the SEP module primary structure as an add-on assembly and will not serve as a part of the SEP module space load carrying structure. The TTA SEP module attachment should be designed to maximize the TTA's adaptability to a wide variety of space vehicle configurations.

4.2.1.2 STRENGTH REQUIREMENT: The TTA structural interface must be capable of maintaining the structural integrity and alignment of the TTA under the launch and flight dynamic environments specified TBD.

4.2.1.3 STIFFNESS REQUIREMENT: The TTA structural interface shall not deflect excessively when subjected to launch or flight loads. In particular the flexibility in the flight (uncaged) configuration shall be compatible with TVC requirements.

4.2.1.4 GROUND HANDLING REQUIREMENT: The TTA should be a structurally self supporting integral unit when removed from the spacecraft and should be capable of withstanding the ground handling forces associated with a 1g gravity field. If ground handling fixtures are required, the TTA must be provided with the necessary handling fixture interface points.

4.2.1.5 INTEGRATION REQUIREMENT: The TTA interface design should allow easy integration with the spacecraft structure and should not place requirements on the spacecraft structure which will result in an excessively heavy spacecraft structure. In general the goal of the interface design should be to minimize the weight of the total spacecraft, not just the weight of the TTA. Interface requirements that can significantly increase the spacecraft structural weight include the following:

- 1) Requirements for negligible relative motion between widely spaced spacecraft interface points.
- 2) Requirements for large numbers of spacecraft interface points, or points in unnatural locations.
- 3) Requirements for a very stiff spacecraft structure.
- 4) Requirements for spacecraft interface points to support moment loads.

4.2.1.6 ATTACHMENT METHOD: The TTA to SEP module attachment should allow easy attachment to, and removal from the SEP module and shall be capable of carrying the necessary interface loads.

#### 4.2.2 POWER CONDITIONER STRUCTURAL INTERFACE

The power conditioner (PC) has a structural interface with the SEP module primary structure and with the thermal louver assemblies. The PC interfaces are controlled by the following factors:

- 4.2.2.1 MOUNTING CONFIGURATION: The PC will mount into a rectangular bay as in integral part of the SEP module primary structure and will serve as a shear plate to prevent parallelogramming of the bay. The thermal louver assemblies will be attached to the PC as an add on subsystem. The PC shall provide the structural interface required by the thermal louver assemblies.
- 4.2.2.2 STRENGTH REQUIREMENT: The PC structure must be capable of maintaining structural integrity and alignment under the launch environments specified in TBD. In its function as a shear plate it must also be designed to carry the shear loads specified by the SEP module structural design.
- 4.2.2.3 STIFFNESS REQUIREMENTS: The PC structure shall not deflect excessively when subjected to the launch environment specified in Document TBD and the shear loads specified by the SEP module structural design. In particular the deflection amplitudes shall be compatible with the requirements of the PC electronic modules and the thermal louver assemblies.
- 4.2.2.4 GROUND HANDLING REQUIREMENT: The PC should be an integral unit when removed from the SEP module and should be capable of withstanding the ground handling forces associated with a 1g gravity field. If ground handling fixtures are required, the PC must be provided with the necessary handling fixture interface points.
- 4.2.2.5 INTEGRATION REQUIREMENT: The PC interface design should allow easy integration with the SEP module structure and should not place requirements on the SEP module structure which will result in an excessively heavy SEP module structure. In general the goal of the interface design should be to minimize the weight of the total SEP module, not just the weight of the PC. Interface requirements that can significantly increase the SEP module structural weight include the following:

- 1) Requirements for negligible relative motion between widely spaced SEP module interface points.
- 2) Requirements for large numbers of SEP module interface points, or points in unnatural locations.
- 3) Requirements for a very stiff SEP module structure.
- 4) Requirements for SEP module interface points to support moment loads.

4.2.2.6      ATTACHMENT METHOD: The PC to SEP module attachment method should allow easy attachment to, and removal from the SEP module and shall be capable of carrying the necessary interface loads. The PC shear plate surface shall accommodate the attachment of a thermal control louver assembly, and as a minimum have a specified flatness, roughness, and fastener pattern.

4.2.3      THRUSTER STRUCTURAL INTERFACE  
(TBD)

4.2.4      PROPELLANT TANK SUPPORT STRUCTURAL INTERFACE  
(TBD)

4.3      FLIGHT DATA SUBSYSTEM (FDS) INTERFACES

4.3.1      GENERAL

The Flight Data Subsystem provides the means for monitoring thrust subsystem performance and for the generation and distribution of sequencing and control commands. The Master FDS is the hub of the Data Subsystem network, distributing all control commands in addition to assembling performance analysis data and action requests (anomalous performance interrupts). The Master FDS (per Fig. 4.3.1) is linked to the thrust subsystem elements via interface control units called FDS slaves. The Master-Slave communications is provided by command buses and data lines. The FDS slave units provide the necessary multiplexing, decoding, and signal conditioning necessary to interface with the thrust subsystem elements.



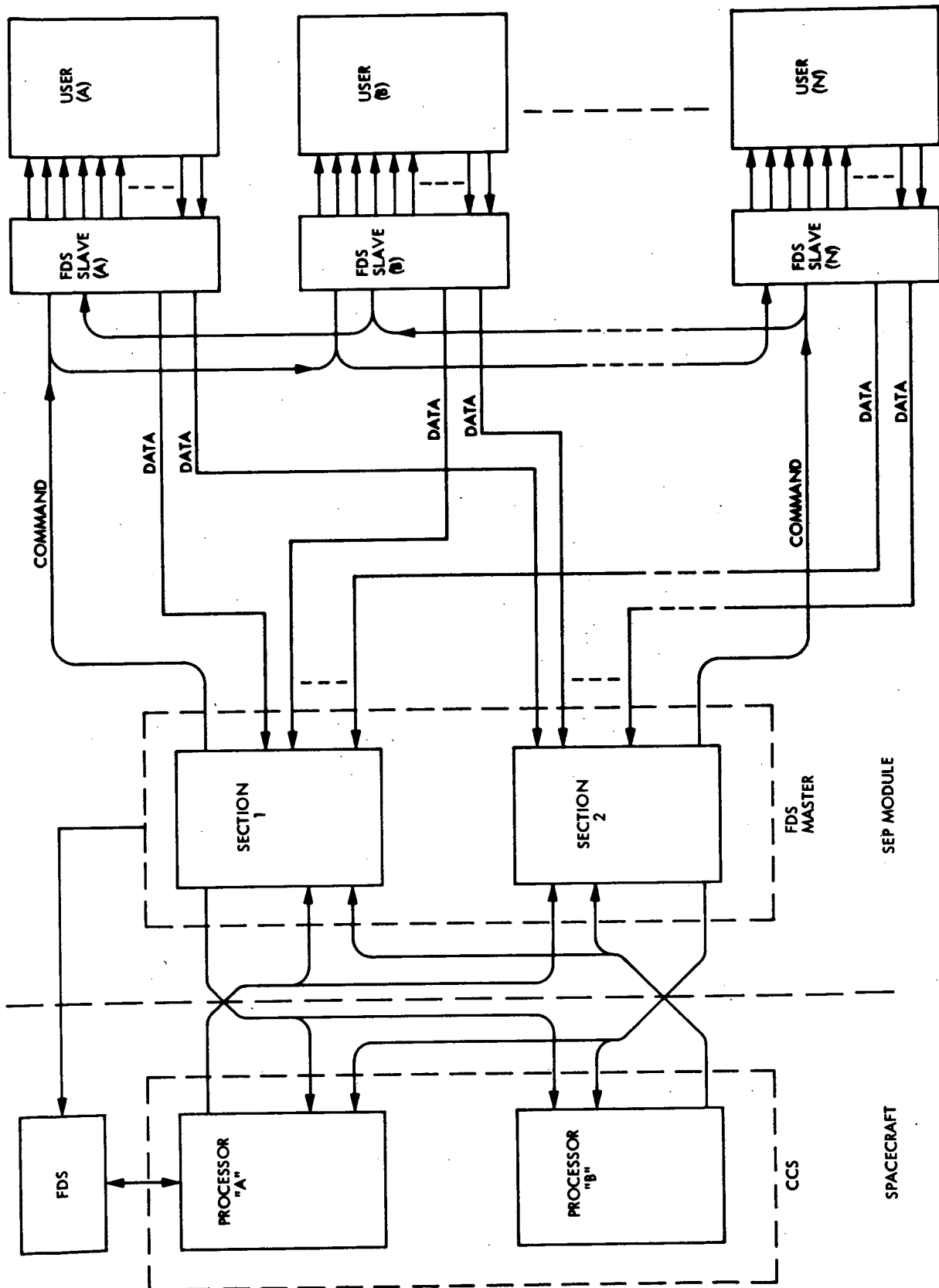


Figure 4.3.1 Data Subsystem Block Diagram

4. 3. 2 INTERFACE SIGNAL CHARACTERISTICS

4. 3. 2. 1 CHARACTERISTICS OF COMMAND SIGNALS AT FDS SLAVE TO USER INTERFACE:

4. 3. 2. 1. 1 Analog Commands:

Resolution of the analog signal will be as required by the user but will not exceed 12 bits plus sign. Accuracy of the analog signal will be related to resolution but will not be better than 0.25% of full scale + 1/2 least significant bit of the resolution. Analog commands will be monotonic within  $\pm 1/4$  least significant bit for resolutions up to and including 8 bits and within  $\pm 1/2$  least significant bit for high resolution.

4. 3. 2. 1. 2 Discrete Digital Commands:

In general, these commands are expected to handle some power, the exact implementation will be dependent will be upon the power and speed requirements of the user.

4. 3. 2. 1. 3 Coded Digital Commands:

These are user dependent and may go to the user in parallel, serial or discrete pulse form.

4. 3. 2. 2 CHARACTERISTICS OF DATA SIGNALS AT THE USER TO FDS SLAVE INTERFACE:

4. 3. 2. 2. 1 Analog Signals:

Analog data are normalized to  $\pm 1.5$  V or 0 to 5 V ranges. The precision of conversion on all analog signals is seven bits. Accuracy of conversion of high level analog signals is 1/2% of reading plus 1/2% of full scale. Temperature measurements are made with resistance thermometers which have a range from 500 ohms to 600 ohms over the temperature range of interest. The accuracy of temperature conversion is 3%. Other transducers may cover the range of 0 to 100 millivolts and the

accuracy of conversion for these is 3%. All analog signals may be high and low limit checked on each scan. Program alert (interrupt) signals are generated by out of limit conditions.

4.3.2.2.2 Discrete Digital Data or Status:

Digital signals are represented by switch closures to ground.

4.3.3 POWER CONDITIONER INTERFACE

The power conditioner provides analog and digital performance data to the FDS slave.

The power conditioners require a variety of power switching discrete commands for start-up, shut down and sequencing. Analog signals are required to establish various current reference levels for control and navigation.

4.3.4 SWITCHING MATRIX INTERFACE

The switching matrix will require power switching discrete commands and will provide digital switch status information.

4.3.5 THRUST VECTOR CONTROL INTERFACE

The TVC will require power switching discrete commands along with positional data (analog or digital). Analog and digital measurement will be required for temperature and position determination.

4.3.6 PROPELLANT TANKAGE AND DISTRIBUTION INTERFACES

Digital discrete commands are required to open and close propellant valves. Temperature and pressure data will be presented in analog form.

4.4 ATTITUDE CONTROL SUBSYSTEM INTERFACE

4.4.1 TVC INTERFACE

A. Electronic Logic

The RCS will supply to the TVC electronic logic information on position.

B. Overrides

The RCS will send a signal to turn off the TVC at any time the RCS requires such as when the RCS deadband is violated. The TVC will remain off until the RCS verifies that the space vehicle position and rate errors are within the normal RCS deadband.

4.5 SEP MODULE THERMAL CONTROL SUBSYSTEM INTERFACE

4.5.1 POWER CONDITIONER INTERFACE

The power conditioner must be designed to accommodate the mounting of a thermal control louver assembly to the PC shear plate.

4.6 SEP MODULE CABLING SUBSYSTEM INTERFACES

4.6.1 ELECTRICAL INTERFACES

The flight Thrust Subsystem electrical cabling connects the interfaces defined in Table 4.6.1.

4.6.2 MECHANICAL INTERFACES

The cabling shall be confined within the thrust subsystem configurational envelope and shall be supported by primary and/or secondary structure and enclosures (i. e., Power Conditioner Compartment) through the use of brackets, clamps and other means. The subsystem cabling group shall develop routing and support requirements to include all mechanical elements which directly support the cables.

Table 4.6.1. Thrust Subsystem Cabling Interfaces

Interface		Connector Description			Circuit Interconnections		Special Types
Origin	Destination	Type	Insert	Quantity	Contacts Used	Spares	
TBD							

5.0 PERFORMANCE PARAMETERS

5.1 SPECIFIC IMPULSE

The specific impulse of the thrust subsystem shall be held constant at 3000 sec  $\pm$  100 sec over the full range of subsystem operating conditions.

5.2 EFFICIENCY

The efficiency of the thrust subsystem, defined as the ratio of the square of the thrust delivered to twice the product of the propellant mass flow rate and the power consumed

[i. e.,  $\eta_{TSS} = F^2 / (2\dot{m}P_{in})$ ], shall be not less than 63% at an input power of 16 kW and a specific impulse of 3000 sec, and shall be not less than 63% at an input power of 1.6 kW and a specific impulse of 3000 sec.

The minimum efficiency shall be a guaranteed efficiency, representing the 3 $\sigma$  lower bound of all uncertainties in the efficiency.

5.3 ALLOCATION OF EFFICIENCY LOSSES

Efficiency losses include  $I^2R$  losses in cables and switches, pointing error losses due to thrust vector control, conversion losses in the power conditioner, and thruster losses. Table TBD gives the allocation of efficiency losses among these elements.

5.4 THRUST CONTROL

The thrust output of the electric thrust subsystem shall be held within  $\pm 5\%$  of its commanded value over the full range of subsystem operating conditions. Of this 5% uncertainty,  $\pm 1\%$  is allocated to pointing uncertainty due to thrust vector control, and  $\pm 4\%$  is allocated to the uncertainties introduced by the thruster and power conditioner operation.

5.5 THRUST VECTOR CONTROL

TBD

5.6

RELIABILITY

The reliability of the electric thrust subsystem shall have a calculated value of greater than 0.96. The reliability required of each subsystem element to meet the overall subsystem reliability is given in Table (TBD).

6.0 PHYSICAL CHARACTERISTICS AND CONSTRAINTS

6.1 MASS CONSTRAINTS

The total dry mass of the electric thrust subsystem shall not exceed 234.4 kg. The table (ref. II-D-1) gives the allocation of subsystem mass among the various subsystem elements.

6.2 CONFIGURATION CONSTRAINTS

The configuration of the thrust subsystem is defined by TBD.

7.0 SAFETY

All designed test activity shall be consistent with requirements stated in TBD.



**DESIGN REQUIREMENT**  
**INFORMATION ONLY**

CODE IDENT NO. \_\_\_\_\_

DES. REQ. TSS - 220 - 1 REV. \_\_\_\_\_

ISSUE DATE June 23, 1972

SUPERSEDING \_\_\_\_\_

DATED \_\_\_\_\_

## APPENDIX B

**DESIGN REQUIREMENT**  
**THRUST SUBSYSTEM      FLIGHT**  
**ELECTRONIC PACKAGING**

\_\_\_\_\_  
**ENGINEER**

**COGNIZANT ENGINEER**

\_\_\_\_\_  
**APPROVED**

**DIVISION 35**  
**REPRESENTATIVE**

\_\_\_\_\_  
**APPROVED**

**Thrust Subsystem**  
**REPRESENTATIVE**

**JET PROPULSION LABORATORY**  
**CALIFORNIA INSTITUTE OF TECHNOLOGY**  
**PASADENA, CALIFORNIA**

**INFORMATION ONLY**

JPL Des Req TSS 200 - 1

**CHANGE INCORPORATION LOG**

CHG LTR	WRITER		AUTHORITY	PAGES AFFECTED	DATE	ENG APPROVAL	
	INITIAL	SECTION				INITIAL	SECTION

# INFORMATION ONLY CONTENTS

<u>Section</u>	<u>Title</u>	<u>Page</u>
1.	SCOPE . . . . .	8
1.1	Scope . . . . .	8
1.2	Applicability. . . . .	8
2.	APPLICABLE DOCUMENTS. . . . .	8
3.	REQUIREMENTS . . . . .	17
3.1	Conflicting requirements . . . . .	17
3.2	Electronic packaging requirements . . . . .	17
3.2.1	General . . . . .	17
3.2.2	Materials, parts, and processes . . . . .	17
3.2.2.1	Materials and processes . . . . .	17
3.2.2.2	Electronic packaging parts and hardware . . . . .	19
3.2.3	Electronic parts . . . . .	19
3.3	System design requirements. . . . .	19
3.3.1	Standard Orbiter Assemblies . . . . .	20
3.3.2	Weight and volume . . . . .	20
3.3.3	Insulation and protection of electrical conductors . . . . .	20
3.3.4	Markings . . . . .	21
3.3.5	High voltage protection . . . . .	21
3.3.6	Contact separation force tests . . . . .	21
3.3.7	Venting . . . . .	21
3.3.8	Conductive particles . . . . .	21
3.3.9	Finishes for electronic equipment Structures . . . . .	21
3.3.9.1	Polymeric coating . . . . .	22
3.3.9.2	Chemical surface treatments . . . . .	22
3.3.9.3	Gold electroplating . . . . .	22
3.3.9.4	Polishing . . . . .	23
3.3.10	Support Equipment (SE) connectors . . . . .	23

**INFORMATION ONLY**

CONTENTS (cont)

<u>Section</u>	<u>Title</u>	<u>Page</u>
3.3.11	Electromechanical interconnections . . . . .	23
3.4	Electronic assembly design . . . . .	23
3.4.1	Electronic assembly geometry . . . . .	24
3.4.2	Electronic assembly (EA) cabling . . . . .	24
3.4.3	Thermal design . . . . .	24
3.4.4	Electronic assembly mechanical design. . . . .	27
3.4.5	Electronic assembly chassis design . . . . .	28
3.4.6	Electronic assembly operational fixture and tooling design . . . . .	28
3.5	Subassembly design . . . . .	28
3.5.1	Subassembly configuration. . . . .	28
3.5.2	Electronic packaging standards . . . . .	28
3.5.2.1	Terminal connected planar packaging. . . . .	29
3.5.2.2	Wire connected modular packaging. . . . .	29
3.5.2.3	Integrated circuit modular packaging. . . . .	29
3.5.3	Subassembly wiring . . . . .	29
3.5.3.1	Wiring design . . . . .	33
3.5.3.2	Mechanical support . . . . .	33
3.5.3.3	Connectors . . . . .	33
3.5.3.4	Wire routing . . . . .	34
3.5.4	Subassembly thermal design . . . . .	34
3.5.5	Subassembly structure . . . . .	35
3.5.6	Sinusoidal vibration . . . . .	35
3.5.7	Shielding . . . . .	35
3.5.8	Subassembly rework . . . . .	36
3.6	Fasteners . . . . .	36
3.6.1	Removable fasteners . . . . .	36
3.6.2	Preferred locking systems for threaded fasteners. . . . .	36
3.6.2.1	Screw in a blind hole . . . . .	36
3.6.2.2	Screw in a through-hole with a nut . . . . .	36

**INFORMATION ONLY****CONTENTS (cont)**

<u>Section</u>	<u>Title</u>	<u>Page</u>
3.6.2.3	Acceptable alternate locking methods . . . . .	37
3.6.3	Threaded fastener torque requirement . . . . .	37
3.6.4	Latent debris control . . . . .	37
3.6.5	Fasteners used in electrical circuits . . . . .	37
3.6.6	Fasteners for mounting connectors . . . . .	37
3.6.7	Nonthreaded fasteners . . . . .	38
3.6.8	Fastener materials . . . . .	38
3.6.9	Special fasteners . . . . .	38
3.6.10	Threaded fasteners . . . . .	38
3.6.10.1	Standard structural fasteners . . . . .	38
3.6.10.2	Packaging fasteners . . . . .	39
3.6.10.3	Washers . . . . .	39
3.7	Polymeric material usage . . . . .	40
3.7.1	Embedment packaging . . . . .	40
3.7.2	Spot bonding . . . . .	40
3.7.3	Conformal coating . . . . .	40
3.7.4	Conductor insulation . . . . .	40
3.7.5	Connector sealing . . . . .	41
3.7.6	Adhesive bonding . . . . .	41
3.7.7	Connector potting . . . . .	41
3.7.8	Nonmetals . . . . .	41
3.7.8.1	Rigid materials used to mold or machine . . . . .	41
3.7.8.2	Polymeric materials . . . . .	42
3.7.8.3	Fire retardant materials . . . . .	42
3.7.8.4	Prohibited materials . . . . .	42
3.7.9	Installation of electronic and mechanical parts . . . . .	43
3.8	Standard fabrication and processes . . . . .	43
3.8.1	General fabrication requirements . . . . .	43
3.8.2	Fabrication environmental process control . . . . .	44
3.8.3	Facility control and cleanliness . . . . .	44
3.8.4	Handling of dangerous materials . . . . .	44
3.8.5	Qualified processes . . . . .	44

**INFORMATION ONLY****CONTENTS (cont)**

<u>Section</u>	<u>Title</u>	<u>Page</u>
3.8.6	Interconnection . . . . .	44
3.9	Material product control . . . . .	45
3.10	Environmental requirements . . . . .	45
3.11	Documentation . . . . .	45
3.11.1	Specifications and procedures . . . . .	45
3.11.2	Materials, parts, and process documentation implementation . . . . .	46
3.11.2.1	JPL in-house . . . . .	46
3.11.2.2	Contractor . . . . .	46
3.11.3	Drawings . . . . .	46
3.11.4	Interface Control Drawings . . . . .	46
3.11.5	Interface definitions . . . . .	46
3.11.6	Installation drawings and procedures . . . . .	46
3.12	Workmanship . . . . .	48
3.13	Rework . . . . .	48
3.14	Integrated circuit module fabrication . . . . .	48
4.	QUALITY ASSURANCE PROVISIONS . . . . .	49
4.1	General . . . . .	49
4.2	Inspection . . . . .	49
5.	PREPARATION FOR DELIVERY . . . . .	49
5.1	Assembly, shipping, and handling equipment . . . . .	49
5.2	Electronic assemblies . . . . .	49
5.3	Battery assembly. . . . .	49
5.4	Subassemblies . . . . .	49
6.	NOTES . . . . .	50
6.1	Definitions . . . . .	50
6.2	Electronic packaging and fabrication documentation . . . . .	51

**INFORMATION ONLY****TABLES**

<u>Number</u>	<u>Title</u>	<u>Page</u>
I	Acceptable CRES Steels . . . . .	18
II	Fastener Diameter and Pitch Combinations . . . . .	39
III	Installation of Electronic and Mechanical Parts . . . . .	43
IV	Electronic Parts Interconnection Techniques . . . . .	45

**ILLUSTRATIONS**

<u>Figure</u>	<u>Title</u>	<u>Page</u>
1.	Typical Integral Shear Plate Chassis Type Electronic Assembly. . . . .	25
2.	Typical Plug-in Subchassis Type Electronic Assembly . . . . .	26
3.	Terminal Connected Planar Packaging . . . . .	30
4.	Wire Connected Modular Packaging . . . . .	31
5.	Integrated Circuit Modular Packaging . . . . .	32
6.	Typical Subassembly Interface Control Drawing . . . . .	47
7.	Electronic Packaging and Cabling Documentation . . . . .	52

**INFORMATION ONLY**

## 1. SCOPE

1.1 Scope. This document covers the electronic packaging design requirements for the TSS electronic equipment. The design requirements for the system, subsystem, assembly, and subassembly are delineated, including design concepts and preferred packaging techniques, materials, processes, and hardware. Included are the geometrical, structural, thermal, weight, volume, and assembly cabling considerations applicable to electronic packaging.

1.2 Applicability. This document is intended to cover the general design techniques to be utilized in the packaging of the TSS electronic equipment. The techniques, methods, materials, and processes specified herein are known to be compatible with TSS requirements. However, in some cases certain requirements may generate special problems requiring design techniques beyond the scope of this document. In those cases, alternate design and process techniques with their associated materials other than those appearing in this document are available. These alternate techniques and materials applicable to electronic packaging shall be qualified and approved prior to use on the TSS in order to assure reliable survival of equipment under mission environmental stress conditions which includes shock, vibration, temperature extremes, and long-term operation in the vacuum of outer space.

## 2. APPLICABLE DOCUMENTS\*

2.1 The following documents form a part of this document to the extent specified herein:

## SPECIFICATIONS

Jet Propulsion Laboratory

ZPH-2239-0940

Precision Hookup Wire, Electrical  
Fluorocarbon Insulated

ZPP-2061-PPL

Preferred Parts List, Reliable  
Electronic Components

ZPP-2063-PMP

Preferred Materials, Fasteners,  
Processes and Packaging and  
Cabling Hardware

\* All documents listed under the heading Jet Propulsion Laboratory are JPL internal documents.



**INFORMATION ONLY**

ZPH-240000	Connector, Electrical Circular, Miniature, Quick Disconnect, Detail Specification for
ZPS-4505-0001	Process Specification, Anodic Treatment of Magnesium Alloys
GMO-20009-PRS	Process Specification, Safe-tieing with Lockwire
GMO-20514-PRS	Process Specification, Installation of Thin Wall Inserts in Aluminum and Magnesium
20045/200	"D" Series, Mark I, Connector (Miniature Type) (Hi-Rel)
20060	Process Specification, Adhesive Coating and Encapsulants for Electronic Packaging
30228	Process Specification, Spacecraft, Flight Equipment, Temperature Control Surfaces Gold Plating (Electrodeposited) for Aluminum and Aluminum Alloys
FS500440	Process Specification, Resistance Welded Electrical Connections for Cross-Wire Opposed Electrodes, Detail Specification for
FS500441	Process Specification, Soldering Process, General Specification for
FS500443	Process Specification, Transformers and Inductors, Electronic Packaging of, Detail Specification for
LS500445	Service Specification, Installation of Electronic Assemblies and Subassemblies, Detail Specification for
TS500446	Part Specification, Connector, Rectangular, Miniature, Contact Separation Force Test, Detail Specification for
FS500447	Process Specification, Intrawiring of Subassemblies, Detail Specification for
ES500448	Equipment Specification, Shipping and Handling Equipment for Electronic Assemblies, Subassemblies and Cables, Detail Specification for
FS500449	Process Specification, Adhesive Bonding, Electronic Packaging and Cabling, Detail Specification for

**INFORMATION ONLY**

LS5004	Service Specification, Handling, Storage and Shipping Requirements for Electronics Assembly, Detail Specification for
FS500451	Process Specification, Identification and Marking Methods for Parts and Assemblies, General Specification for
LS500452	Service Specification, Connector Installation, Rectangular Miniature, Detail Specification for
FS500627	Process Specification, Buffing Aluminum Alloys, Detail Specification for
FS500628	Process Specification, Buffing Magnesium Alloys and Electroplated Magnesium Alloys, Detail Specification for
FS501424	Process Specification, Application of Temperature Control Paints, Detail Specification for
FS501437	Process Specification, Spacecraft Flight Equipment, Temperature Control Surfaces and Certain Electronics Packaging Components Gold Plating (Electrodeposited) for Magnesium Alloy, Detail Requirements for
BS502673	Material Specification, Permanent Marking Ink (Wornowink, Series M/Catalyst A), Detail Specification for
BS502674	Material Specification, Permanent Marking Colors (Cat-L-Ink, Series 50-000/20), Detail Specification for
FS502705	Process Specification, Conversion Coating (Dow 19) Magnesium Alloys, Detail Specification for
FS502754	Process Specification, Conversion Coating (Dow7) of Magnesium Alloys, Detail Specification for
LS504255	Service Specification, Torque Requirements, Fasteners, Structural, Spacecraft, and Electronic Equipment, Detail Specification for
FS504270	Process Specification, Impact Installation of Thin Wall Threaded Inserts
FS505920	Process Specification, Preparation for, Installation and Inspection of Solid Rivets and Screws, Detail Specification for

**INFORMATION ONLY**

\*TS.

Environmental Test Specification, Type Approval and Flight Acceptance Test Requirements and Preacceptance Test Limits (assembly Level)

General Speci-

fication for

FS506079

Manufacturing Process Specification, Printed Wiring Boards and Assemblies, Detail Specification for

Federal

QQ-W-343

Wire, Electrical and Nonelectrical Copper (Uninsulated)

Military

MIL-S-7742

Screw Threads, Standard, Optimum Selected Series, General Specification for

MIL-A-8625

Anodic Coatings for Aluminum and Aluminum Alloys

MIL-G-45204

Gold Plating, Electrodeposited

**SYSTEM DOCUMENTS**

Jet Propulsion Laboratory

Design Criteria

Characteristics and Restraints

**REQUIREMENTS**

**INFORMATION ONLY** JPL Des Req TSS - 220 - 1

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TSS -2009-1	Design Requirement, TSS Flight Equipment, System Cabling, Cable, and Harness Assemblies
DM505139	Design Requirement, High Voltage Elec- tronic Packaging, Flight Equipment
DM505140	Design Requirement, Modular Integrated Circuit, Electronic Packaging, Flight Equipment
DM505141	Design Requirement, Wire Connected Modular, Electronic Packaging, Flight Equipment
DM505142	Design Requirement, Planar Electronic Packaging, Flight Equipment

STANDARD

Federal

FED-STD-209

Clean Room and Work Station Require-  
ments, Controlled Environment

**INFORMATION ONLY** JPL Des Req TSS - 220 - 1

## PROCESS BULLETINS

### Jet Propulsion Laboratory

ZBE-1020-0001	Process Bulletin, Electronic Subassembly Equipment Storage and Handling Container
ZBE-1041-0007	Process Bulletin, Bonding of Printed Wiring, Terminal and Insulation Boards with Eccobond 55/9
ZBE-1041-0008	Process Bulletin, Spot Bonding of Component Parts with Stycast 1095/9
ZBE-1041-0009	Process Bulletin, Spot Bonding, Electronic Parts with Solithane 113/300 Filled Polyurethane
ZBE-1051-0003	Process Bulletin, Installation of Temperature Transducer
ZBE-1051-0004	Process Bulletin, Insulation and Sealing of Subminiature Rectangular Miniature and Circular Quick Disconnect Connectors
ZBE-1061-0004	Process Bulletin, Integrated Circuit Module Insulation Coating

**INFORMATION ONLY**

ZBE-1061-0005	Process Bulletin, Modification and Rework of Planar Electronic Subassemblies
ZBE-1061-0006	Process Bulletin, Application of Solithane 113/300 Conformal Coating (Fluorescent and Nonfluorescent)
ZBE-1061-0007	Process Bulletin, Stress Relief Coating of Embedded Components
ZBE-1071-0003	Process Bulletin, Embedment of Electronic Modules Using Stycast 1090/11
ZBE-1072-0003	Process Bulletin, Conformal Encapsulation of Electrical Conductors
ZBE-1072-0004	Process Bulletin, Coating of Electrical Conductors Exposed to Ultraviolet Radiation
ZBE-1073-0002	Process Bulletin, Impregnation and Embedment of Transformers and Inductors
ZBE-1074-0001	Process Bulletin, Potting of Subminiature Rectangular Connectors
ZBE-1074-0002	Process Bulletin, Potting of Circular Connectors
ZBE-1081-0006	Process Bulletin, Hand Soldering Stranded Wire into Electrical Connector Contact Solder Cups
ZBE-1081-0007	Process Bulletin, Magnet Wire Routing and Soldering IC Module
ZBE-1091-0001	Process Bulletin, Fabrication and Use of Lacing Ties
ZBE-1092-0001	Process Bulletin, Splicing of Electrical Hookup Wire
ZBE-1092-0002	Process Bulletin, Shielding Terminations
ZBE-1102-0001	Process Bulletin, Component Installation IC Module
ZBE-1103-0001	Process Bulletin, Preparation and Installation of Terminal Integrated Circuit (IC Module)

**INFORMATION ONLY**

## DRAWINGS

Jet Propulsion Laboratory

ST10062	Insert, Threaded, Self-Locking, Thin Wall, Capped.
ST10063	Insert, Threaded, Self-Locking, Thin Wall
ST10066	Wire, Magnet, Copper, Solder-Through Insulation
ST10638	Hookup Wire, Extruded, Teflon, Type E, 600 Volt Rating, 200°C
ST10646	Screw, Quint Lock
ST11478	Wire, Hookup, Teflon, Single Conductor
ST11479	Cable, Electric, Hookup, Teflon, Single Conductor, Shielded and Unjacketed
ST11480	Cable, Electric, Hookup, Teflon, Twisted Pair
ST11481	Cable, Electric, Hookup, Teflon, Twisted pair, Shielded and Jacketed (TPSJ)
ST11482	Cable, Electric, Hookup, Teflon, Twisted Triad
ST11483	Cable, Electric, Hookup, Teflon, Twisted Triad, Shielded and Jacketed
ST11484	Cable, Electric, Hookup, Teflon, Twisted Quad
90127	Transipad Installation
90303	Transipad Installation
90413	Connector Pattern, Pygmy and PT00 Back Wall Mounting
115838	Removal Tool, Cannon Connectors
4901045	Subchassis Outline
10000857	Forging Chassis
10006879	Baseline Configuration, Half Size Subchassis
10006880	Baseline Configuration, Full Size Subchassis
10017435	IC Module Subassembly
10017463	IC Module Locating Pin

**INFORMATION ONLY** JPL Des Req TSS - 220 - 1

## PUBLICATIONS

### Air Force Technical Order

TO-00-25-203

Standards and Guidelines for the Design  
and Operation of Clean Rooms and  
Supplemental Devices

### American Society for Testing and Materials

D 351-62

Standard Specification for Natural  
Muscovite Mica Based on Visual Quality

F50-65T

Continuous Counting and Sizing of Air-  
borne Particles in Dust-Controlled Areas  
By the Light-Scattering Principle (For  
Electronic and Similar Applications)

(Copies of specifications, standards, procedures, drawings, and publications required by suppliers in connection with specific procurement functions should be obtained from the procuring activity or as directed by such activity.)



**INFORMATION ONLY**

### 3. REQUIREMENTS

3.1 Conflicting requirements. In case of conflict between the requirements of this document and the requirements of any document referenced herein, the conflict shall be referred to the procuring activity or contracting officer for resolution.

### 3.2 Electronic packaging requirements.

3.2.1 General. The electronic packaging of electronic space flight equipment shall be in accordance with the requirements specified herein. The packaging shall be compatible with applicable electronic requirements and shall meet the launch and flight environmental requirements. Protection during fabrication, bench handling, testing, shipping, and storage shall be included. The design shall meet the requirements of long-life operation and survival in space environment. Access shall be provided to any area of the equipment that may require adjustment, repair, or modification.

### 3.2.2 Materials, parts, and processes.

3.2.2.1 Materials and processes. Materials and processes used for electronic equipment shall be selected from JPL Spec ZPP - 2063 - PMP Processes for electronic equipment and the exact issue invoked

The requirements within the scope of JPL SPEC ZPP - 2063 - PMP shall govern the selection of the type and class of materials used for electronic packaging design.

3.2.2.1.1 Polymerization. All polymers, such as conformal coatings and embedment materials applied to the electronic equipment, shall be cured to a specified standard condition. Certification or engineering documentation shall be required specifying the procedures used to effect the polymerization.

**INFORMATION ONLY**

3.2.2.1.2 Qualification. All polymeric materials used to fabricate electronic equipment shall be selected from JPL SPEC ZPP - 2063 - PMP. Qualification of other material for use in the electronic equipment shall be based on the specific intended application of each material and controlled by JPL-approved documentation.

3.2.2.1.3 Metals. The partial listing of metals taken from JPL SPEC ZPP - 2063 - PMP and listed herein apply to the specific applications for the electronic packaging uses as defined herein.

3.2.2.1.3.1 Magnesium. Magnesium parts shall be fabricated from AZ31B-F plate of ZK60A-T5 forgings as listed in JPL SPEC ZPP - 2063 - PMP. Magnesium forgings used to fabricate electronic equipment chassis and subchassis, and other related critical machined magnesium components, shall meet the requirements of JPL Drawing 10000857.

3.2.2.1.3.2 Aluminum. Parts fabricated from aluminum shall conform to JPL SPEC ZPP - 2063 - PMP. Aluminum parts requiring a polished surface shall be fabricated from 6061 alloy.

3.2.2.1.3.3 Stainless steels. Corrosion-resistant (CRES) steels used shall be nonmagnetic. The preferred material is A286 alloy in accordance with JPL SPEC ZPP - 2063 - PMP. The 300 series austenitic alloys (alternate), in accordance with JPL SPEC ZPP - 2063 - PMP, are acceptable provided they are annealed and passivated after fabrication (Table I).

Table I. Acceptable CRES Steels

Sequence	Type
Preferred	A286
First alternate	316 or 310
Second alternate	302, 303, or 304

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3.2.2.1.3.4 Brass. All brass used for parts such as terminals, contacts, or connectors shall be manufactured from free cutting brass (composition 22) or cartridge brass per JPL SPEC ZPP - 2063 - PMP.

3.2.2.2 Electronic packaging parts and hardware. Electronic packaging parts and hardware used in electronic equipment shall meet the requirements of 3.5.2 and 3.6, and shall be selected from JPL SPEC ZPP - 2063 - PMP.

3.2.3 Electronic parts. Electronic parts shall be selected from JPL SPEC ZPP - 2061 - PPL.

3.3 System design requirements. The electronic packaging aspects of the system design shall conform to the requirements of the following JPL Documents:

DM505139

DM505140

DM505141

DM505142

TSS electronic equipment shall be designed to contain removable and replaceable electronic assemblies. Assemblies shall generally be of standardized size and shape to provide system design flexibility. Assemblies shall contain standardized subassembly chassis to allow for interchangeability. Location of assemblies shall balance the considerations of the center of gravity, temperature control, system cabling, and other significant system requirements. The general arrangement of electronic equipment assemblies and associated subassemblies shall be controlled by JPL Drawing.

**INFORMATION ONLY**

3.3.1 Standard assemblies. TSS electronic equipment shall be designed to be contained in removable and replaceable electronic assemblies. Assemblies shall be of standardized size and shape to provide system design flexibility. Assemblies shall contain standardized sub-assemblies to allow for interchangeability and to meet the requirements of the overall system.

3.3.2 Weight and volume. Electronic equipment requiring a completely new electronic packaging design shall be designed for minimum weight that will provide the required volume and satisfy the requirements of reliability, ease of fabrication, environmental stress, handling, durability, operational considerations, and flexibility to permit modification or rework without significant degradation. Proposed designs previously used on shorter duration space missions must meet these requirements and shall be carefully evaluated to determine compliance with mission requirements.

3.3.3 Insulation and protection of electrical conductors. Electrical conductors, except the engagement surfaces of connector contacts, shall be completely insulated to prevent short circuits resulting from contact with conductive foreign material during operation of the TSS. In cases where it is not practical to coat conductors due to functional requirements, physical isolation shall be provided to prevent short circuits. The following preferred techniques shall be used per 3.7.4 to satisfy these requirements:

- a. Install shrinkable sleeving to connector solder cups and connector inserts. The base of the connector solder cups shall be insulated.
- b. Conformally coat all exposed electrical conductors and electronic parts.
- c. Conformally encapsulate electrical conductors, such as solder joint projections and other sharp edges that are not completely insulated by conformally coating.

**INFORMATION ONLY**

3.3.4 Marking. Assembly and subassembly markings shall be in accordance with JPLSPEC ZPP - 2063 - PMP , and as shown on Figures 1, 2, 3, and 5 herein. Markings shall be of a contrasting color, yellow or black, in accordance with the requirements specified in JPL Specification FS500451, using ink per JPL Specification BS502673 or BS502674. Location of the subassembly identification shall be visible when the subassembly is installed within a chassis. Modified subassemblies shall be marked in accordance with JPL Process Bulletin ZBE-1061-0005.

3.3.5 High voltage protection. All electronic equipment using voltages in excess of 250 volts dc or peak ac shall meet the requirements of JPL Design Requirement DM505139 to prevent corona or arcing through the pressure range of Earth ambient to space vacuum. The applicability of this requirement shall include electronic equipment that utilize high voltage only after passing through the critical pressure region.

3.3.6 Contact separation force tests. "D" series connectors (with socket contacts) shall have separation force tests per JPL Specification TS500446. The tests shall be conducted on all "D" series connectors just prior to (each) anticipated final engagement of mating connectors.

3.3.7 Venting. Equipment containing sealed spaces or voids that can trap or contain gases or liquids shall be adequately vented, or properly designed to operate and maintain the required pressure in the space vacuum for a minimum period of three years.

3.3.8 Conductive particles. When equipment is inaccessible for cleaning after assembly, the design shall provide for control or containment of conductive particles resulting from fastener installation.

3.3.9 Finishes for electronic equipment structures. Faying surfaces between the electronic assemblies and electronic subassemblies shall have finishes that will provide electrical grounding and efficient thermal paths to assist in temperature control ; and to assure continuity of the electronic equipment chassis ground circuit when consistent with detail functional requirements. In

**INFORMATION ONLY**

general, finishes for electronic equipment structures within the structure shall have high absorptance and emittance to promote heat transfer. The following requirements within the scope of JPL SPEC ZPP - 2063 - PMP shall govern the treatments, platings, or application of coatings to produce qualified surface finishes.

3.3.9.1 Polymeric coating. Polymeric coating material, color white, PV-100, Vita Var Paint Company (or equal) is required for electronic equipment thermal surfaces. Application process shall be per JPL Specification FS501424.

3.3.9.2 Chemical surface treatments. The preferred subchassis finish is a dichromate applied over a grit blasted surface per JPL Specification FS502754. Within the scope of JPL SPEC ZPP - 2063 - PMP, the following treatments for electronic equipment surfaces, as required, shall be used:

- a. Aluminum anodized (nonconductive). The anodizing process shall be specified as Type III, Class 1 of MIL-A-8625.
- b. Dichromate treatment for magnesium (conductive). The dichromate treatments shall be Dow No. 7 (or equal) per JPL Specification FS502754.
- c. Anodize treatment for magnesium (nonconductive). The anodizing treatment shall be Dow No. 17 (or equal) per JPL Specification ZPS-4505-0001.
- d. Touchup for Dow No. 7 and Dow No. 17 rework. The touchup or rework treatment for damaged or scratched Dow No. 7 and Dow No. 17 treated surfaces shall be the application of Dow No. 19 per JPL Specification FS502705.

3.3.9.3 Gold electroplating. Gold electroplating for electronic equipment surfaces, as required, shall be in accordance with the following requirements:

- a. Aluminum. The process for plating aluminum shall be per JPL Specification 30228 (gold).

**INFORMATION ONLY**

- b. Magnesium. The process for plating magnesium shall be per JPL Specification FS501437.

3.3.9.4 Polishing. Polishing for electronic equipment surfaces, as required, shall be in accordance with the following requirements:

- a. Aluminum. The process for polishing aluminum shall be as specified in JPL Specification FS500627.
- b. Gold-plated magnesium. The process for polishing gold-plated magnesium shall be per JPL Specification FS500628.

3.3.10 Support equipment (SE) connectors. Separate connectors shall be provided on the electronic assembly cable harnesses for the inter-connection of all SE test cables used to support the system and subsystem level testing. Normal flight cabling shall not be disturbed for SE testing purposes.

3.3.11 Electromechanical interconnections. Mechanical interconnections designed to perform electrical functions shall be covered by JPL-approved processes, and shall satisfy the necessary performance requirements in a repetitive and reliable manner.

The completed interconnections, except for mechanical connections on approved connectors, shall be visually examined with documented acceptance criteria. Interconnections shall be provided with protection from damage resulting from stresses that occur from assembly, test, or operation on those parts designed to be removable from, or that move during operation or use.

3.4 Electronic assembly design. The electronic packaging design concept shall require that electronic assemblies within the TSS structure form an integral part of the TSS structure. Subassembly structures shall be mechanically integrated into the assembly chassis and the assembly chassis, in turn, shall be fastened to the TSS structure. The outer shear plates shall be utilized as load-bearing members. Two types of standard assembly

INFORMATION ONLY

designs shall be utilized for the electronic assemblies on the TSS, except for the thruster power conditioner and switching matrix assembly,

which shall utilize special design configurations. Electronic assembly structural characteristics shall satisfy the requirements of JPL Functional Requirement.

3.4.1 Electronic assembly geometry. The configuration of one type of electronic assembly for use within the standard integrated shear plate chassis shall conform to Figure 1. The other type of electronic assembly shall utilize a standard plug-in equipment chassis configuration and a separate shear plate. This configuration shall conform to Figure 2. Within an assembly, the subassembly profile shall be standardized to facilitate location flexibility. To optimize design and to reduce interconnections, standards shall be utilized per Figures 1 and 2. Subassembly thickness may be varied as necessary to satisfy the particular equipment design requirements and packaging techniques. The Relay Radio/Telemetry and Pyro Control Assembly shall utilize a variation of the standard integrated shear plate chassis design.

3.4.2 Electronic assembly (EA) cabling. Electronic assembly cabling shall be securely fastened to supports by the use of Ty-Raps, cable ties, cable straps, or cable clamps, etc., to prevent cable damage or malfunction during handling, testing, and flight. Connector mounting shall assure straight and free engagement of contacts, and all "D" series connectors shall be unmated using appropriate tooling. The plug-in equipment electronic assembly design requires that the "D" series connectors mounted on the chassis float radially for proper alignment to subassembly connectors. Direct access connectors shall be covered by protective caps when not mated. Cabling design shall be in accordance with the requirements specified in JPL Design Requirement TSS -2009-1. The cabling configurations for the two EA types are shown on Figures 1 and 2.

3.4.3 Thermal design. The electronic assemblies shall be designed and fastened to provide conductive heat paths to the thermal control surfaces consistent with special equipment requirements

Subassemblies



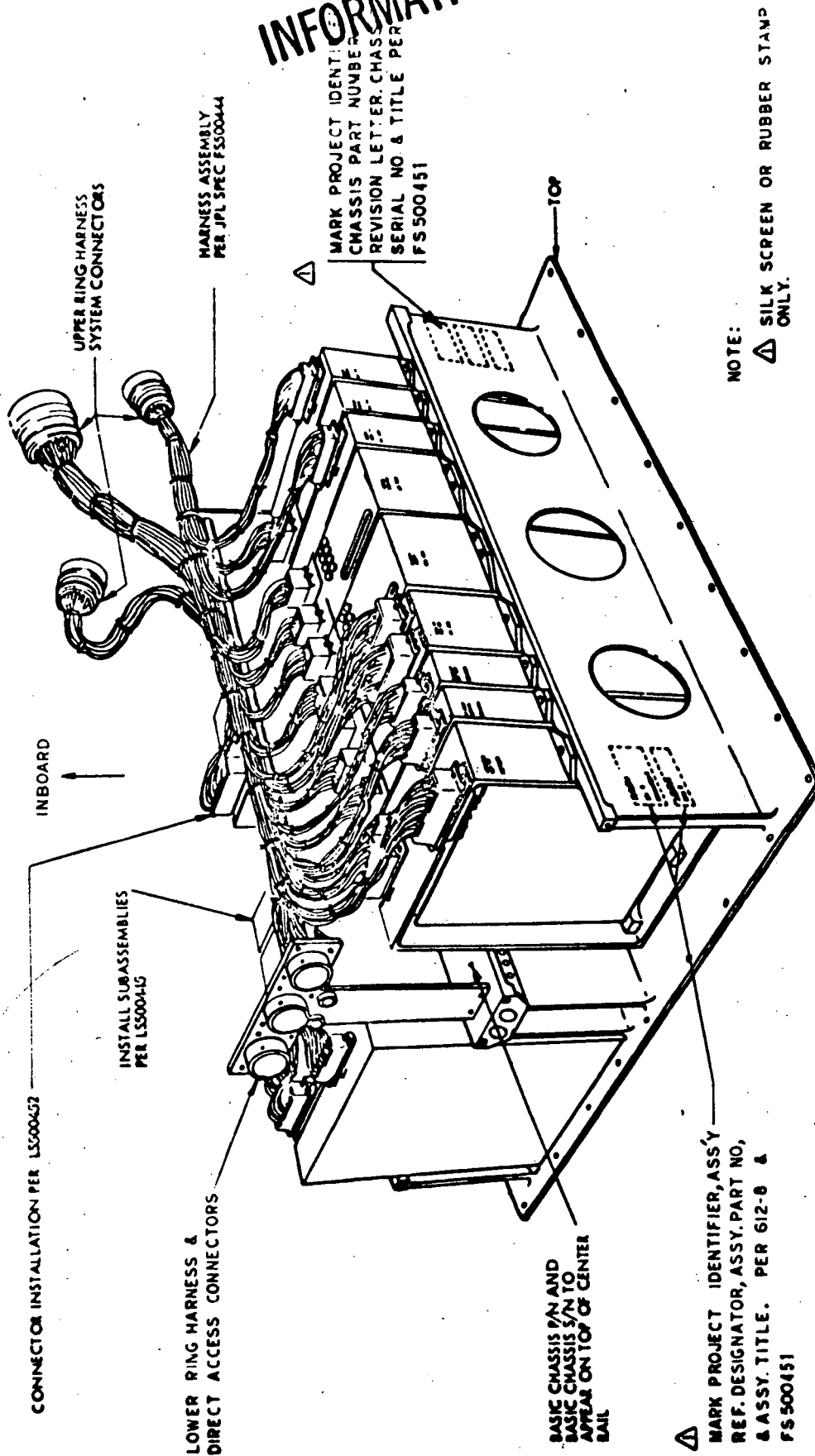
**INFORMATION ONLY**

Figure 1. Typical Integral Shear Plate Chassis Type Electronic Assembly

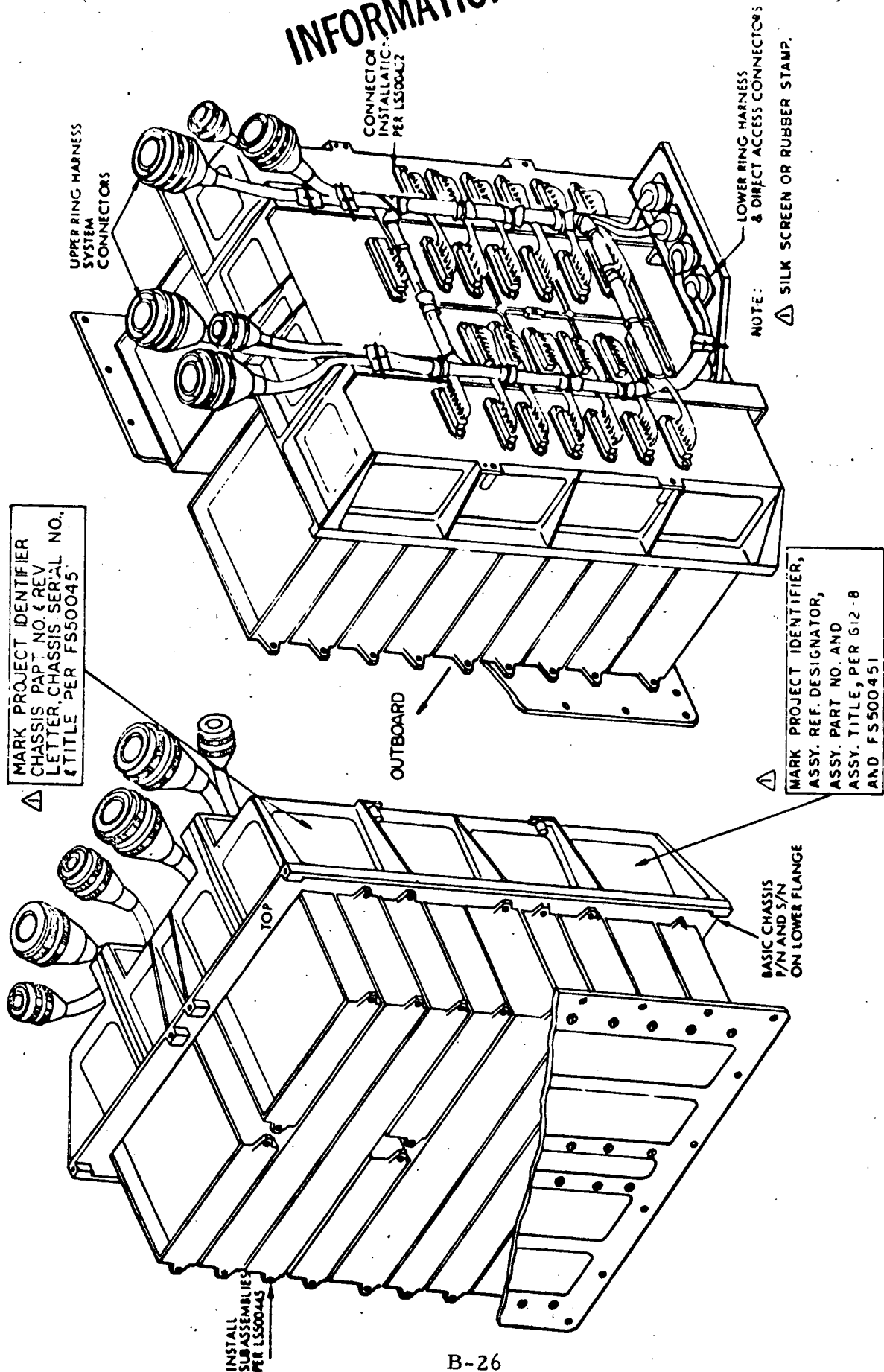


Figure 2. Typical Plug-in Subchassis Type Electronic Assembly

**INFORMATION ONLY**

within each assembly shall be capable of serving as low-impedance thermal conductors. Subassemblies with high heat dissipation shall be distributed to assist in the temperature control of all elements within an assembly. Adjacent subassemblies shall utilize radiative and conductive heat transfer between themselves to the maximum extent consistent with other requirements. The basic assembly chassis shall provide a surface suitable for application of required thermal control finishes. Surface flatness and the quantity of fasteners used shall be compatible with the thermal design and dissipation requirements. The design shall assure that the operating temperature of the electronic parts shall not exceed their derated values in the worst case thermal conditions of operation and environments. Typically, the thermal impedance of a single joint between a Dow No. 7 surface finished subchassis and a Dow No. 17 finished chassis is 3°F/watt when a 6-32 screw is used. The thermal impedance of a typical joint assembled with a 8-32 fastener is 2°F/watt.

3.4.4 Electronic assembly mechanical design. The design of the mechanical elements of electronic equipment shall assure minimum thermal and mechanical stress imposed on electronic parts within the total range of environments. Assembly chassis structures shall have structural characteristics so that the resulting dynamic environment imposed upon electronic subassemblies is compatible with all the approved electronic packaging techniques utilized. The outboard surface elements of all assemblies shall be designed to function as primary structural shear plates mounted on the structure; all holes, discontinuities, and load attachments in this outboard shear plate shall be controlled in order to assure structural integrity. Subassemblies shall be designed as assembly load-sharing members and mechanically integrated into the assembly chassis structure. electronic assemblies shall be designed to isolate the electronic subassemblies from the stress and strain of the structure. In order to adequately decouple from the launch vehicle and the structure resonant frequencies during launch, the primary resonant frequency of electronic assemblies shall be a minimum of 260 Hz in any axis. The fundamental resonant frequency of subassemblies in any axis shall be greater than 400 Hz in order to achieve additional mechanical decoupling and to control the surfaces immediately related to component parts and their

**INFORMATION ONLY**

associated interconnections. In general, with an input vibration level of 5 g's, the gains at resonant frequencies should be less than 30 measured at discrete component parts relative to the octagonal structure/electronic assembly interfaces.

3.4.5 Electronic assembly chassis design. The structural design of the EA chassis, as a minimum, shall meet the applicable requirements of JPL Functional Requirement.

The assembly chassis structures shall have characteristics so that the resulting shock, vibration, and thermal environments are compatible with the electronic parts and electronic packaging techniques utilized. Fabrication shall conform to the requirements shown on the applicable drawings.

3.4.6 Electronic assembly operational fixture and tooling design. The operational fixtures and tooling shall be designed and qualified to permit reliable installation and removal of assemblies with no resultant damage or degradation. Electronic subassemblies and assemblies shall be installed in accordance with applicable specifications.

3.5 Subassembly design.

3.5.1 Subassembly configuration. The configuration of the subassembly shall be standardized for each of the two assembly designs utilized, unless restricted by electronic design or by special sensor or component configurations. The standardized subchassis geometry shall be in accordance with JPL Drawings 4901045, 10006879, and 10006880.

3.5.2 Electronic packaging standards. Electronic equipment shall utilize one or a combination of the electronic packaging standards specified in 3.5.2.1 through 3.5.2.3. These standards are approved for electronic packaging. The use of other electronic packaging techniques shall require prior project approval by the packaging engineer.

**INFORMATION ONLY**

3.5.2.1 Terminal connected planar packaging. Terminal connected planar packaging is shown on Figure 3. This method shall utilize printed wiring and terminal boards for mounting and interconnecting the components. This method is versatile and provides the maximum circuit and component part change capability. The technique is suitable for power supplies and other circuits requiring component sizes that are incompatible with modular packaging. Design requirements of this technique shall be as specified in JPL Design Requirement DM505142. The preferred plating material for printed wiring board is gold (per JPL Specification FS506079).

The incorporation of shielded compartments and point-to-point layout of components for RF equipment shall be considered a variation of the planar terminal connected packaging technique.

3.5.2.2 Wire connected modular packaging. Wire connected modular packaging shall be as shown on Figure 4. This method utilizes welded cordwood modules interconnected with a soldered wire harness matrix. The technique provides reliable high-density packaging, ease of fabrication and module replacement, and permits changes in the subassembly interconnections without resultant degradation. Design requirements for this technique is specified in JPL Design Requirement DM505141.

3.5.2.3 Integrated circuit modular packaging. Integrated circuit modular packaging shall be as shown on Figure 5. This method utilizes integrated circuit flat packs welded in modules and interconnected with soldered magnet wire. Discrete components may be used as required by soldering to bifurcated terminals. The modules are interconnected with a subassembly by the use of parallel wire groups with soldered connections. Design requirements for this technique are specified in JPL Design Requirement DM505140.

3.5.3 Subassembly wiring. Subassembly wiring shall meet the requirements of JPL Specification FS500447. Wire shall be tape wrapped, or precision extruded; all wires shall be screened.

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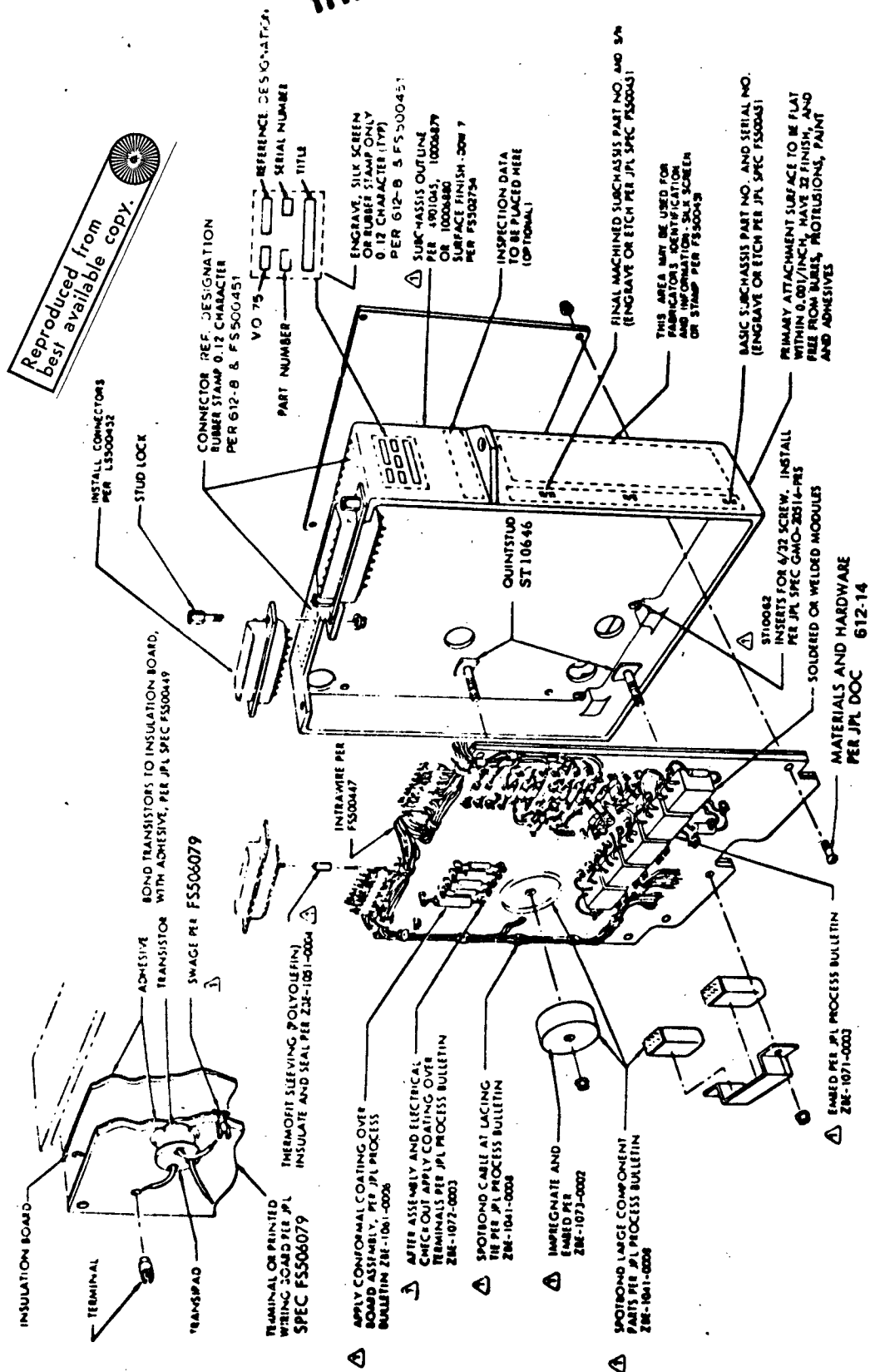


Figure 3. Terminal Connected Planar Packaging



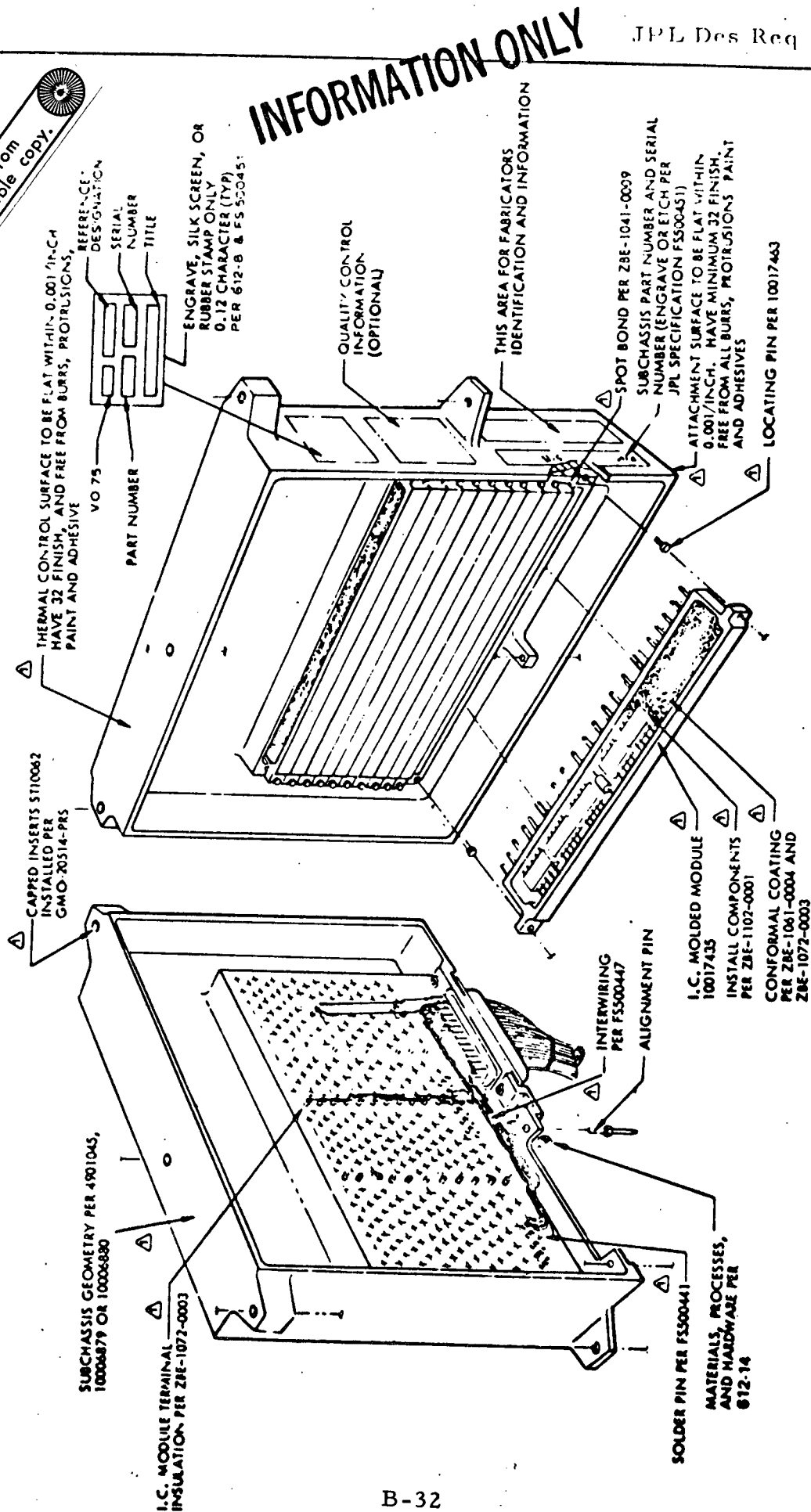


Figure 5. Integrated Circuit Modular Packaging



**INFORMATION ONLY**

3.5.3.1 Wire design. The subassembly design shall allow adequate space for routing wires to prevent the wire insulation from touching the corners of terminals or other parts that may have a detrimental effect upon the wire insulation.

3.5.3.1.1 Stranded hookup wire. Stranded hookup wire used for electronic equipment intraconnections shall be per JPL Specification ZPH-2239-0940 and JPL Drawings ST10638 or ST11478 through ST11484. The insulation shall be 600 volt, tape wrapped, or precision extruded, unpigmented, bondable, virgin TFE.

3.5.3.1.2 Shielded and jacketed hookup wire. Shielded and jacketed hookup wire shall be per JPL Specification ZPH-2239-0940 and JPL Drawings ST11478 through ST11484. The jacket insulation shall be 600 volt, tape wrapped, or precision extruded, unpigmented, bondable, virgin TFE. The shielded primary conductors shall meet the requirements specified in 3.5.3.1.1.

3.5.3.1.3 Twisted pairs, triad and quad hookup wire. Twisted hookup wire shall be per JPL Specification ZPH-2239-0940 and JPL Drawings ST11478 through ST11484. The insulation shall be 600 volt, tape wrapped, or precision extruded, bondable, virgin TFE.

3.5.3.1.4 Uninsulated solid hookup wire. Uninsulated solid hookup wire shall be Type S, copper, tinned, and annealed per QQ-W-343.

3.5.3.1.5 Insulated solid hookup wire. Insulated solid hookup wire used to fabricate IC modules shall be per JPL Drawing ST10066.

3.5.3.2 Mechanical support. Wires (especially at terminations) shall be supported by potting, clamping, spot tying, or spot bonding as applicable to prevent degradation as a result of installation, test, or environmental stress.

3.5.3.3 Connectors. The connectors in subassemblies shall be sealed and insulated per 3.7.5. Connectors shall meet the requirements of the following paragraphs.

**INFORMATION ONLY**

3.5.3.3.1 Rectangular. Rectangular connectors are listed in JPL  
The rectangular connectors shall meet the requirements of JPL Specification 20045/200.

3.5.3.3.2 Circular. Circular connectors and allowable configurations are listed in JPL  
The circular connectors shall meet the requirements of JPL Specification ZPH-2245-0300.

3.5.3.3.3 Other connectors. Other connectors, such as high voltage, RF, umbilical, and the flight quality mating SE connectors, shall be specifically identified and controlled by detail engineering documents.

3.5.3.4 Wire routing. The wiring shall be routed to minimize critical lead lengths and possible degradation of electrical performance.

3.5.3.4.1 Direct access functions. As a design objective, the grouping of direct access functions shall be in an individual connector so that a direct access harness can service the subsystem and be removed prior to flight.

3.5.4 Subassembly thermal design. The subassembly packaging design shall permit the component parts to operate within the temperature limits for which they have been qualified. The design shall impose minimum thermal stress to the component parts, and shall meet the thermal gradient requirements between components and the temperature control surfaces. The outboard base of the electronic subassemblies (opposite to the electrical connectors) shall be designed to serve as primary heat conducting surfaces. Significant heat producing components shall be located as close as practical to this surface. The subchassis web shall be the primary heat sink for component parts. The subassembly design shall control the thermal impedance between the components and the subchassis web, including the thermal characteristics of the subchassis structure, as required, to permit electronic parts to operate within their qualified

**INFORMATION ONLY**

temperature limits for the worse case of temperature environment and heat dissipation. The thermal impedance for 6-32 screws tying the subassembly to the chassis shear plate is 3°F/watt; for 8-32 screws, the impedance is 2°F/watt.

3.5.5 Subassembly structure. The subassembly structures (sub-chassis) shall be designed as mechanically integrated electronic assembly load-bearing members. The subassembly structures shall be of adequate stiffness to assure that component fragility levels are not exceeded. Subassemblies shall be designed to assure that components, modules, and interconnecting devices are not damaged by deflections caused by the shock and vibration environments. The minimum subassembly primary resonant frequency in any axis shall be 400 Hz.

3.5.6 Sinusoidal vibration. Subassemblies shall be designed to withstand the following sinusoidal vibration levels measured at their attachment points. These levels shall apply in any direction:

<u>Frequency</u> <u>(Hz)</u>	<u>Level</u> <u>(g rms)</u>
5 - 10	0.4 inch double amplitude
10 - 40	2.0
40 - 100	10.0
100 - 2000	20.0

3.5.7 Shielding. Electronic subassemblies shall be shielded, as required, to prevent detrimental effects from externally or internally generated electromagnetic (or electrostatic) interference.

**INFORMATION ONLY**

3.5.8 Subassembly rework. Subassembly rework shall be performed to complete engineering documentation including part removal and replacement. The procedures used shall not damage or degrade the status of the reworked unit. Embedded modules for flight equipment shall not be reworked. When modification or repair of equipment designed to meet the requirements specified in JPL Design Requirement DM505142 is required, the disassembly of solder joints, the replacement of polymeric materials, and the reidentification shall be in accordance with JPL Process Bulletin ZBE-1061-0005.

3.6 Fasteners. Fasteners shall be selected from those listed in JPL SPEC ZPP - 2063 - PMP. The fasteners selected shall satisfy the requirements of the following paragraphs.

3.6.1 Removable fasteners. All threaded fasteners shall utilize a method of locking to prevent loosening of the fasteners when the equipment to which the fastener is used is subjected to the specified environment. The locking method shall be identifiable and inspected for compliance with the locking requirements.

3.6.2 Preferred locking systems for threaded fasteners.

3.6.2.1 Screw in a blind hole. Fasteners for blind hole locking shall be accomplished by using one of the following methods:

- a. Locking thin wall insert per JPL Drawing ST10062 or ST10063, with Long-Lok screw.
- b. Nonlocking press nut with Nylock CRES screw.

3.6.2.2 Screw in a through-hole with a nut. Fasteners with a through-hole shall be provided with an A286 CRES or 300 CRES screw and a self-locking nut, HW 42 series (Kaynar Manufacturing Company).

**INFORMATION ONLY**

3.6.2.3 Acceptable alternate locking methods. The acceptable alternate locking methods listed is arranged in descending order of preference. The use of these locking methods shall be limited to subassemblies. The locking methods listed shall not be used on subsystem interfaces.

- a. Lockwire, per JPL Specification GMO-20009-PRS.
- b. Secure with approved spot bonding material, exterior surface of nut-to-screw thread.

3.6.3 Threaded fastener torque requirement. All threaded fasteners shall be tightened to a torque value per JPL Specification LS504255.

**CAUTION**

To avoid damage when mounting an electronic part within a subassembly, the fastening torque shall be based primarily upon the degree of fragility of the part. The applicable JPL drawing or the part manufacturer's recommendation shall have precedence over any of the general torque requirements specified in JPL Specification LS504255.

3.6.4 Latent debris control. In areas where the complete removal of metal chips and shavings (a condition which could exist during fastener installation) cannot be assured, capped nuts or capped inserts shall be used during the assembly of parts requiring threaded fasteners.

3.6.5 Fasteners used in electrical circuits. Fasteners shall not be used within the subassemblies as part of an electrical circuit.

3.6.6 Fasteners for mounting connectors. Fasteners for mounting the following types of connectors shall be in accordance with the following requirements:

**INFORMATION ONLY**

- a. D series. Fasteners for "D" series connectors shall be installed and torqued as specified in JPL Specifications LS500452 and LS504255. "D" series mounting hole patterns and hardware shall be as specified in JPL Specification LS500452.
- b. Pygmy, Type DS311. The mounting hole requirements for type DS311 shall be as specified in JPL Drawing 90413.

3.6.7 Nonthreaded fasteners. Nonthreaded fasteners (rivets) shall conform to the following requirements:

- a. Rivet holes. Rivet holes shall be in accordance with JPL Specification FS505920.
- b. Rivet inspection. Rivet inspection shall be in accordance with JPL Specification FS505920.

3.6.8 Fastener materials. All fastener hardware shall be constructed from materials that are corrosion-resistant and nonmagnetic. The A286 steel, annealed and passivated 300 series stainless steel, or specified brass and polymers shall be acceptable.

3.6.9 Special fasteners. Requirements for special fasteners are recognized and will be accepted in the interests of effective design and functional performance. However, the use of special fasteners shall be minimized and shall be qualified and controlled by complete engineering documentation.

3.6.10 Threaded fasteners.

3.6.10.1 Standard structural fasteners. The standard structural fasteners used to attach components to the primary structure shall be selected from JPL ZPP - 2063 - PMP.

**INFORMATION ONLY**

3.6.10.2 Packaging fasteners. Fasteners used on electrical equipment shall be selected from JPL SPEC ZPP - 2063 - PMP. Where fasteners utilize a locking insert, the fastener screw or bolt shall have an integral rectangular locking plastic insert in accordance with the applicable JPL standard drawing. In addition to the locking requirements, alternate or special fasteners shall conform to the following:

- a. Diameter and thread pitch combinations. Fasteners shall be limited to a selection from the diameter and pitch combinations shown in Table II. Grip lengths shall be in 1/16-inch increments.
- b. Thread form and class. Packaging fasteners shall have a thread form per MIL-S-7742.

Table II. Fastener Diameter and Pitch Combinations

Diameter and Pitch	Requirement
0 - 80 UNF 2 - 56 UNC 4 - 40 UNC 6 - 32 UNC 8 - 32 UNC 10 - 32 UNF 1/4 - 28 UNF 5/16 - 24 UNF	Use Federal Thread Standard Handbook, H-28, for threads larger than 5/16-24 UNF.

3.6.10.3 Washers. Star washers shall not be used. Washers may be used under JPL screw part number DS136 for subassembly installation in electronic assemblies and other structures when required. Flat washers may be used to distribute stresses over a larger area when required. Washers shall not be used in lieu of the self-locking hardware requirements contained in this design requirement.

**INFORMATION ONLY**

3.7 Polymeric material usage. The application of polymeric materials, in accordance with the processes of 3.7.1 through 3.7.7, shall provide electrical insulation, mechanical support, and heat transfer necessary to meet the general electronic packaging requirements.

3.7.1 Embedment packaging. Embedment packaging of electronic equipment, as required, shall be performed in accordance with the following requirements:

- a. Modules and electronic parts, other than transformers, solenoids, and inductors shall, when specified, be embedded in accordance with JPL Process Bulletin ZBE-1071-0003.
- b. Transformers, solenoids, and inductors shall be terminated, embedded, and encapsulated in accordance with JPL Specification FS500443 and JPL Process Bulletin ZBE-1073-0002.

3.7.2 Spot bonding. Electronic parts, wire bundles, modules, and fasteners, where external locking is specified, shall be secured in accordance with JPL Process Bulletin ZBE-1041-0008 or ZBE-1041-0009.

3.7.3 Conformal coating. All electronic parts, part leads, printed wiring, and other current carrying conductors shall be protected and insulated in accordance with JPL Process Bulletin ZBE-1061-0006. For complete electrical insulation, further conductor insulation is required per 3.7.4.

3.7.4 Conductor insulation. The insulation of conductors shall conform to the requirements of the following:

- a. Protected areas. Part leads, soldered and welded joints, and other electrically conductive areas not sufficiently covered when coated per JPL Process Bulletin ZBE-1061-0006, shall be insulated in accordance with JPL Process Bulletin ZBE-1072-0003.
- b. Exposed areas. Parts leads, solder and weld joints, and other electrical conductors exposed to ultraviolet



**INFORMATION ONLY**

radiation shall be insulated in accordance with JPL Process Bulletin ZBE-1072-0004.

3.7.5      Connector sealing.      The uninsulated portion of the connector solder cups between the connector insert and the base of the shrinkable tubing shall be sealed and insulated in accordance with JPL Process Bulletin ZBE-1051-0004.

3.7.6      Adhesive bonding.      Adhesive bonding and the application of foam materials shall be in accordance with JPL Specifications 20060 and FS500449.

3.7.7      Connector potting.      Harness and other cabling connector terminations shall be protected from handling damage in accordance with JPL Process Bulletins ZBE-1074-0001 and ZBE-1074-0002.

3.7.8      Nonmetals.

3.7.8.1      Rigid materials used to mold or machine.      Sheet, bar, rod, and other rigid materials used to mold or machine nonmetallic parts shall be in accordance with the following requirements:

- a.      Epoxy glass laminates.      Epoxy glass laminates for printed wiring, terminal, and insulation boards shall conform to JPL Specification FS506079. Laminates for fabricating multilayer boards using plated-through holes for interlayer electrical connections, shall be evaluated by JPL, or the subcontractor considering the specific application and the manufacturing process controls available.
- b.      Acetal.      Parts made from acetal shall be fabricated from Delrin 100 series or Delrin 500 series.
- c.      Fluorocarbon.      Fluorocarbon parts fabricated from film or sheet shall be Teflon FEP, Type A. Molded parts shall be fabricated from Teflon FEP, Type 100 or TFE 7. Machined parts shall be fabricated from TFE 7.

**INFORMATION ONLY**

- d. Polyester. Polyester parts fabricated from film or sheet shall be Mylar, Type A or Type T.
- e. Mica. Insulators used between electronic parts and metal surfaces shall be made from clear natural mica, Grade V-1 clear, per ASTM D 351-62.

3.7.8.2 Polymeric materials. Within the scope of JPLSPEC ZPP -2063 - PMP and using the documents listed therein, the specifications and bulletins for processing of the polymeric materials for use in packaging electronic equipment shall be controlled by JPL manufacturing process specifications and bulletins as specified in 3.7 through 3.7.7. Application of polymeric materials to electronic equipment without qualified processes shall be unacceptable.

3.7.8.3 Fire retardant materials. Addition of materials to polymeric compounds, such as transformer encapsulants to achieve a fire retardant system shall be unacceptable.

3.7.8.4 Prohibited materials. The following materials listed shall be prohibited for use in the electronic packaging. This prohibition shall be applied to the use of pure or commercially pure elements, and to alloys or surface films containing more than 50 percent by weight of these elements, unless specifically qualified.

- |               |  |
|---------------|--|
| a. Arsenic    | o. Acrylonitrile butadiene styrenes (ABS plastics) |
| b. Bromine    | p. Cellulose acetate butyrates                     |
| c. Cadmium    | q. Cellulose nitrates                              |
| d. Cesium     | r. Isoprene rubbers                                |
| e. Iodine     | s. Natural rubbers                                 |
| f. Mercury    | t. Neoprene rubbers                                |
| g. Phosphorus | u. Polystyrenes                                    |
| h. Polonium   | v. Polysulfide rubbers                             |
| i. Potassium  | w. Polyvinyl alcohols                              |
| j. Rubidium   | x. Polyvinyl chlorides                             |
| k. Sodium     | y. Polyvinylidene chlorides                        |
| l. Sulfur     | z. Rubber hydrochloride films                      |
| m. Tellurium  | aa. Styrene butadiene rubbers                      |
| n. Zinc       |  |

**INFORMATION ONLY**

3.7.9 Installation of electronic and mechanical parts. The installation, mounting, and fastening of electronic and mechanical parts and assemblies shall be in accordance with the requirements of Table III.

Table III. Installation of Electronic and Mechanical Parts

Part Description	JPL Specification/Bulletin
Inserts, thin wall	GMO-20514-PRS (non-impact) FS504270 (impact)
Printed wiring, terminals and insulation board laminates	ZBE-1041-0007
Temperature transducers	ZBE-1051-0003
Lacing ties	ZBE-1091-0001
Active electronic parts	DM505142
Bifurcated, swaged terminals, DS series	FS506079
Solder joints	FS500441 ZBE-1081-0006
Subassemblies	LS500445
Connector installation	LS500452 JPL Drawing 90413
Connector removal	JPL Drawing 115838
Connector contact separation force test	TS500446
Transistor insulators (transipads)	JPL Drawing 90127 JPL Drawing 90303

### 3.8 Standard fabrication and processes.

3.8.1 General fabrication requirements. The fabrication of equipment shall be defined and controlled by formal engineering documentation.

INFORMATION ONLY

3.8.2 Fabrication environmental process control. Fail safe temperature and vacuum sensing and protective devices shall be incorporated in a redundant manner in all environmental facilities used in the fabrication and processing of flight hardware. Continuous surveillance of equipment during processing shall be provided when redundant controls are not provided. The process temperature should minimize the thermal stress on parts during fabrication. The process temperature shall not exceed the burn-in temperature of the screened electronic parts used. The lowest part temperature shall not exceed that specified for Flight Acceptance (FA) or that specified in the applicable detail specification.

3.8.3 Facility control and cleanliness. All facilities used for electronic assembly shall be certified as a class 100,000, or better, clean rooms in accordance with the methods and criteria of FED-STD-209, ASTM F-50-65T, and Air Force Technical Order TO-00-25-203.

Personnel access to this area shall be limited to personnel directly performing, monitoring, or supporting assembly operations, and supporting clean room operations. Personnel in the area shall wear suitable clean room garments, such as head covering, gloves, and gown of nonshedding material. Light intensity shall be a minimum of 100 foot-candles on the work surface.

3.8.4 Handling of dangerous materials. Recognized safety practices and instructions of the manufacturer shall be followed in the handling of flammable or toxic materials.

3.8.5 Qualified processes. The processes and procedures for use on electronic equipment shall be in accordance with JPL SPEC ZPP -2063- PMP.

3.8.6 Interconnection. Interconnection techniques of electronic parts are defined by JPL documents listed in Table IV.

INFORMATION ONLY

3.9 Material product control. The formulation, mixing, and repacking of materials shall be documented and controlled, using JPL Form 1888, or JPL-approved equivalent.

3.10 Environmental requirements. The electronic equipment shall be capable of successfully operating in the environmental conditions (as applicable) as specified in the FA and TA testing per JPL Specification TS506000.

3.11 Documentation. The electronic equipment shall be documented,

Table IV. Electronic Parts Interconnection Techniques

Interconnection Techniques	JPL Specification/Bulletin
Soldered connections	FS500441 ZBE-1081-0006
Welded interconnections	FS500440
Etched conductors (planar design)	FS506079
Cabling	ZBE-1092-0001 ZBE-1092-0002 FS500447 ZBE-1061-0004 ZBE-1103-0001

3.11.1 Specifications and procedures. Specifications and procedures shall be prepared per JPL Control Document as necessary, to control, supplement, and integrate the drawings, design, and fabrication processes and methods.

**INFORMATION ONLY**

3.11.2 Materials, parts, and process document implementation.

3.11.2.1 JPL in-house. The appropriate document shall be specified in JPL specifications and on the applicable drawings, as necessary, to assure complete definition and product control. In many cases additional information shall be necessary to aid in the proper electronic packaging or cabling design and document application.

3.11.2.2 Contractor. Implementation of these documents at a contractor's facility shall be per JPL Control Document.

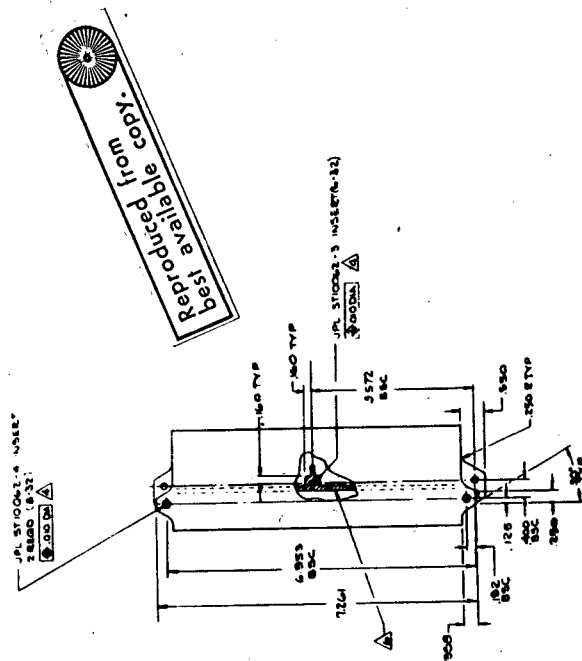
3.11.3 Drawings. All drawings of electronic equipment and hardware shall be prepared in accordance with the requirements specified in JPL Control Document.

3.11.4 Interface Control Drawings. Interface Control Drawings shall be prepared by JPL for each item identified by a functional reference designation when documented and fabricated by a contractor for JPL.

Additional information pertinent to the interface as well as the weight and the center of gravity may also be required and should be coordinated with the electronic packaging technical representative. Figure 6 shows a typical electronic subassembly Interface Control Drawing.

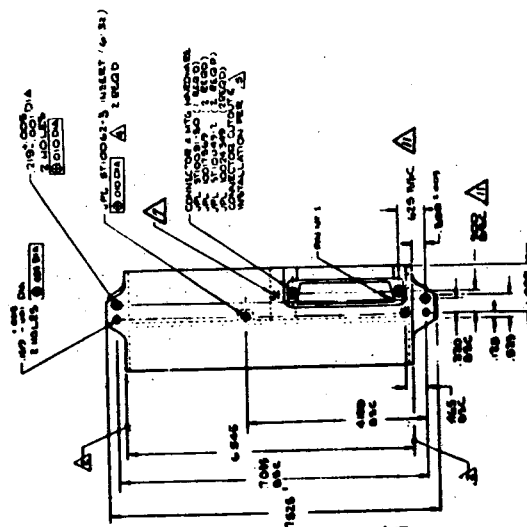
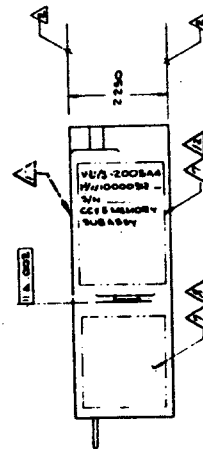
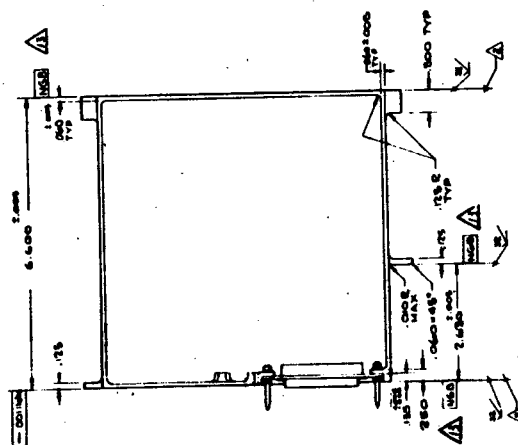
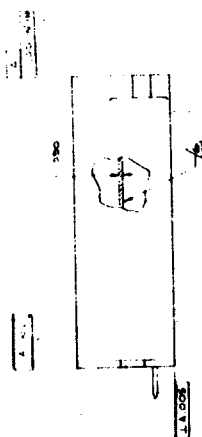
3.11.5 Interface definitions. The control drawings per 3.11.4 and other interface documents for electronic equipment shall be generated and maintained as specified in JPL Control Document. Interface areas included shall be the mechanical and cabling interface definition and control between electronic assemblies and the primary and secondary structural assemblies, and between electronic subassemblies and the associated electronic assemblies.

3.11.6 Installation drawings and procedures. Installation drawings and procedures shall be generated by JPL for all electronics equipment and cabling.



1. ~~1~~ 200 REMOTES NO GRV BLAST.  
 2. ~~1~~ 100000000 75% REFERENCE DESIGNATION SUB-  
 ASSEMBLY, 100000000 WITH APPROPRIATE DESIGN LETTER, 100000000  
 3. ~~1~~ 100000000 THESE DIMENSIONS & TOLERANCES APPLY ONLY TO THE HOLES  
 PATTERN AND ARE NOT APPLICABLE TO INSTALLED CONNECTORS.  
 4. ~~1~~ 100000000 SUBASSEMBLY MATERIAL TO BE AS-TO ORDERED PER MIL-STD-883C.  
 5. ~~1~~ 100000000 REQUIREMENT PER MIL-STD-883C.  
 6. ~~1~~ 100000000 CONTRACTOR INFORMATION MAY APPEAR IN THIS AREA  
 7. ~~1~~ 100000000 RUBBER STAMP OR SILKSCREEN 12 HIGH CHARACTERS  
 8. ~~1~~ 100000000 PER JPL SPEC 3300451  
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NOTES: UNLESS OTHERWISE SPECIFIED



**B-47**

Figure 6. Typical Subassembly Interface Control Drawing

**INFORMATION ONLY**

3.12 Workmanship. Workmanship shall be a uniformly high quality. There shall be no cracks, breaks, chips, bend, burrs, loose attaching parts, or any other evidence of poor workmanship. Uniformity of shapes, dimensions, and construction shall permit interchangeability of replaceable components and complete units. Markings shall remain legible after unit assembly. The unit shall be clean and free of foreign materials. Workmanship shall be in accordance with the requirements of the applicable detail specifications.

3.13 Rework. When modification or repair of equipment design is required to meet the requirements specified in JPL Design Requirement DM505142, the disassembly of solder joints, the replacement of polymeric materials, and the reidentification shall be in accordance with JPL Process Bulletin ZBE-1061-0005.

3.14 Integrated circuit module fabrication. The following JPL process bulletins shall apply to the fabrication of integrated circuit (IC) modules:

- a. ZBE-1081-0007
- b. ZBE-1102-0001



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#### 4. QUALITY ASSURANCE PROVISIONS

4.1 General. Detail design, fabrication, and test shall be subject to the provisions of the approved quality and reliability plans meeting the requirements of JPL Control Documents.

4.2 Inspection. Flight configuration articles shall be inspected to a JPL-approved inspection procedure, or procedures, that satisfactorily examine and display all critical features of the equipment specified herein.

#### 5. PREPARATION FOR DELIVERY

5.1 Assembly, shipping, and handling equipment. The electronic equipment shall be protected against damage, contamination, or general degradation of the equipment during fabrication, handling, assembly, testing, and transportation. The transportation of electronic equipment shall be in conformance with JPL Specification ES500448.

5.2 Electronic assemblies. All electronic assemblies shall be mounted to their respective fixtures per JPL Specification LS500450. Assemblies and flight chassis shall use the handling fixtures and containers (or equivalent) as defined in JPL Specification ES500448 for all phases of assembly and handling, when they are not installed on a test fixture or on the TSS. The assemblies in fixtures placed into the inner container may be used for hand carrying the assembly. The inner container shall be placed into the electronic assembly shipping outer container in accordance with the requirements specified in JPL Specification LS500450, prior to any other mode of transportation.

5.3 Battery assembly. Handling and transportation of the battery assembly shall be in accordance with JPL Specification LS500450.

5.4 Subassemblies. Handling and storage of subassemblies shall meet the requirements specified in JPL Process Bulletin ZBE-1020-0001. The transportation of electronic subassemblies shall be accomplished by using the handling frames and containers (or equal) as defined on the applicable JPL drawing, as applicable during fabrication, testing, and inspection.

**INFORMATION ONLY**

## 6. NOTES

6.1 Definitions. Definitions (as applicable to this document) are as follows:

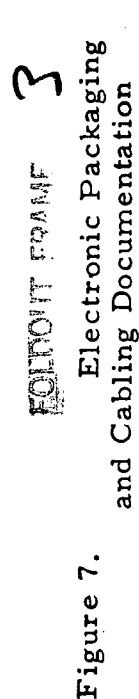
- a. Adhesive. A plastic film used between flat surfaces to bond them together.
- b. Chassis. The major structure for an electronic assembly.
- c. Component part. A part of a subassembly such as a bracket, resistor, capacitor, or casting.
- d. Conformal coating. A thin continuous resin film applied over electronic parts and electrical conductors to provide electrical insulation, moisture protection, support of small lead mounted component parts, and viscoelastic damping during vibration.
- e. Electronic packaging. The conversion of electronic circuits into hardware including the mechanical integration into the system. Also included is the consideration of such factors as mechanical design, thermal design, fabrication techniques and processes.
- f. Encapsulation. Complete covering of a component resulting in different exterior shape and size.
- g. Electronic assembly (EA). A removable major functional electronic unit that may contain an entire subsystem, elements of more than one subsystem, or only a portion of a subsystem.
- h. Electronic subassembly. A functional, testable, and removable unit of an electronic assembly.
- i. Module. A prepackaged unit of a subassembly usually treated as an electronic part, such as a flip-flop, gate, etc.
- j. Spot bonding. Use of plastic material to secure or stabilize parts where a thick localized application of adhesive material is required.

- INFORMATION ONLY**
- k. Subclass. The structure for a subassembly.
  - l. Subsystem. The aggregate of subassemblies and other hardware elements comprising a particular functional whole such as Modulation Demodulation, Relay Telemetry, Radio, etc. Subsystem is a functional class and may include one or several major pieces of equipment.

6.2 Electronic packaging and fabrication documentation. The documentation system for the design and fabrication of the electronic packaging for electronic equipment is shown on Figure 7.

DO NOT PHOTOGRAPH

INFORMATION ONLY



JPL Technical Memorandum 33-583, Vol. II  
DESIGN REQUIREMENT

INFORMATION ONLY

CODE IDENT NO. \_\_\_\_\_

DES. REQ. TSS -2009-1 REV. A

ISSUE DATE June 28, 1972

SUPERSEDING \_\_\_\_\_

DATED \_\_\_\_\_

APPENDIX C

FLIGHT EQUIPMENT  
THRUST SUB-SYSTEM CABLING  
CABLE AND HARNESS ASSEMBLIES

ENGINEER \_\_\_\_\_ DATE \_\_\_\_\_  
Cognizant Engineer

APPROVED BY \_\_\_\_\_ DATE \_\_\_\_\_  
Cabling Group  
Supervisor

APPROVED BY \_\_\_\_\_ DATE \_\_\_\_\_  
Manager, Section 357

APPROVED BY \_\_\_\_\_ DATE \_\_\_\_\_  
Division 35,  
Project Engineer

APPROVED BY \_\_\_\_\_ DATE \_\_\_\_\_  
JET PROPULSION LABORATORY Division 38,  
CALIFORNIA INSTITUTE OF TECHNOLOGY Project Engineer  
PASADENA, CALIFORNIA

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JPL Des Req TSS-2009-1 A

**CHANGE INCORPORATION LOG**

CHG LTR	WRITER		AUTHORITY	PAGES AFFECTED	DATE	ENG APPROVAL	
	INITIAL	SECTION				INITIAL	SECTION
A			Flight		6-28-72	JCA	357

# INFORMATION ONLY CONTENTS

<u>Section</u>	<u>Title</u>	<u>Page</u>
1.	SCOPE . . . . .	5
2.	APPLICABLE DOCUMENTS . . . . .	5
3.	REQUIREMENTS . . . . .	7
3.1	Conflicting requirements . . . . .	7
3.2	General . . . . .	7
3.3	Applicability . . . . .	7
3.4	Materials, parts, and processes . . . . .	7
3.5	Design . . . . .	8
3.5.1	Design objectives . . . . .	8
3.5.1.1	Weight and size . . . . .	8
3.5.2	General restraints . . . . .	8
3.5.2.1	Electrical . . . . .	8
3.5.2.2	Mechanical . . . . .	8
3.5.2.3	Thermal . . . . .	9
3.5.3	Specific restraints . . . . .	9
3.5.3.1	Interchangeability . . . . .	9
3.5.3.2	Conductor size and current capacity . . . . .	10
3.5.3.3	Circuit segregation . . . . .	10
3.5.3.4	Interconnections . . . . .	10
3.5.3.4.1	Connectors . . . . .	10
3.5.3.4.2	Splices . . . . .	10
3.5.3.5	Connector pin assignment . . . . .	11
3.5.3.6	Circuit and cable assignment . . . . .	12
3.5.3.7	High voltages . . . . .	13
3.5.4	Environmental . . . . .	13
3.5.5	Operational . . . . .	13



# **INFORMATION ONLY**

## **CONTENTS (cont)**

<u>Section</u>	<u>Title</u>	<u>Page</u>
3.5.5.1	Warnings .....	13
3.5.5.2	Connector mating and demating .....	13
3.5.5.3	Installation and removal of harnesses .....	13
3.6	RF circuits .....	13
3.7	Fabrication .....	14
3.8	Reliability .....	14
3.9	Pyrotechnic firing circuit cables and harnesses ...	14
3.10	Identification .....	14
3.11	Configuration .....	14
3.12	Electrical test criteria .....	14
3.13	Workmanship .....	14
4.	QUALITY ASSURANCE PROVISIONS .....	15
4.1	Inspections and tests .....	15
4.2	Rejection and resubmittal .....	15
5.	PREPARATION FOR DELIVERY .....	15
6.	NOTES .....	15

## **TABLE**

<u>Number</u>	<u>Title</u>	<u>Page</u>
I.	Current Carrying Ratings (Continuous Duty) of Wires and Cables (TFE Jacket) .....	9

**INFORMATION ONLY****1. SCOPE**

1.1 This document covers the design requirements for Thrust Subsystem flight cable and harness assemblies.

**2. APPLICABLE DOCUMENTS**

2.1 The following documents, of the issue specified in the contractual instrument, form a part of this document to the extent specified herein. For JPL internal use, the issue shall be as specified by the JPL cognizant engineer.

**SPECIFICATIONS\*\*****Jet Propulsion Laboratory**

FS500444	Manufacturing Process Specification, Cabling Fabrication and Assembly, Spacecraft Flight Equipment, General Specification for
ES500448	Equipment Specification, Shipping and Handling Equipment for Electronic Assemblies, Subassemblies and Cables
FS500451	Process Specification, Identification and Marking Methods for Parts and Assemblies, General Specification for
*TS	Environmental Test Specification, Thrust Subsystem General Assembly Level Type Approval and Flight Acceptance
ZPP-2063-PMP	Environmental Test Specification Preferred Materials, Fasteners, Processes, and Packaging and Cabling Hardware

**REQUIREMENTS\*\*****Jet Propulsion Laboratory**

* TSS-3-100	Thrust Subsystem , Characteristics and Restraints
* TSS-3-230	Functional Requirement, Thrust Subsystem , Equipment and Weight List
* TSS-3-260	Functional Requirement, Thrust Subsystem , Electrical Interface and Grounding
* TSS-4-2009	Functional Requirement, Thrust Subsystem , Cabling

\* Not released as of the date of release of this document.

\*\*JPL internal documents.

**INFORMATION ONLY**  
TSS- JPL Des Req TSS-2009-1 A

Design Requirement, Thrust Subsystem  
Flight Equipment, Electronic Packaging

DM505139

Design Requirement, High Voltage  
Protection

DRAWINGS\*\*

Jet Propulsion Laboratory

\* (TBD)

\* (TBD)

Circuit Data Sheet Index and Guide

Thrust Subsystem Flight Cable Interconnect  
Diagram

\*Not released as of the date of release of this document.  
\*\*JPL internal documents.

## SAFETY MANUAL

INFORMATION ONLY

Air Force Eastern Test RangeAFETRM 127-1  
1 January 1969

Range Safety Manual, Volume 1

(Copies of specifications, standards, procedures, drawings, and publications required by suppliers in connection with specific procurement functions should be obtained from the procuring activity or as directed by such activity.)

### 3. REQUIREMENTS

3.1 Conflicting requirements. In case of conflict between the requirements of this specification and the requirements of any document referenced herein, the conflict shall be referred to the procuring activity or cognizant negotiator for resolution.

3.2 General. The requirements specified herein conform to the requirements of JPL Functional Requirement TSS-4-2009. Cable and harness assemblies designed, fabricated and furnished under this design requirement shall be tested and pass the tests specified in Section 4 of JPL Specification FS500444.

3.3 Applicability. The requirements stated herein apply to all types of cable and harness assemblies used in flight TSS's. Additional requirements, applicable to individual assemblies shall be as specified in the applicable interface drawing or circuit data sheet.

3.4 Materials, parts, and processes. Unless otherwise specified on the applicable drawing, materials, parts, and processes used on the assemblies shall be in conformance with JPL Spec ZP<sup>P</sup>-2063-PMP.

3.5 Design. The following design criteria shall be in effect.

3.5.1 Design objectives. The primary design objectives of the cable and harness design shall be to fulfill the requirements of the subsystem wiring interfaces.

3.5.1.1 Weight and size. The design shall emphasize the application of components and techniques that result in the minimum size and weight, compatible with subsystem and reliability requirements and shall meet the applicable weight requirements of JPL Functional Requirement TSS-3-230.

3.5.2 General restraints. The following criteria shall govern general restraints.

3.5.2.1 Electrical. The cable harnesses shall conduct the power and signals in a manner compatible with the requirements of the source unit and the destination unit. Prime considerations shall include:

- a. Control of magnetic and electrostatic interference.
- b. Voltage drop in conductors.
- c. Voltage drop in grounding connections.
- d. Insulation electrical characteristics.
- e. Reduction of subsystem power losses.

3.5.2.2 Mechanical. Mechanical aspects of cable harnesses include insulation strength (tensile and bearing), tolerance to vibration, adequate support, accessibility during Orbiter construction, test and rework. In the physical installation, an attempt shall be made to:

- a. Avoid areas where adverse conditions may exist, or if unavoidable, adequate protection shall be provided.
- b. Avoid interference with optical instruments. Test connector location shall provide for ease of access and preclude damage to any instruments when test harnesses are connected.

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3.5.2.3 Thermal. Harness design, especially harnesses which must flex during flight, shall consider thermal cycling effects caused by alternate Sun and shade exposure. Proper heat protection and dissipation shall be provided. The current carrying capacity of wires and connectors shall be derated to meet anticipated combinations of thermal and vacuum conditions of space, launch, and injection. De-rated current values are presented in Table I.

Table I. Current Carrying Ratings (Continuous Duty)  
of Wires and Cables (TFE Jacket)

Wire Size (AWG)	Continuous-Duty Current (amperes)	
	Single-Wire In Free Air	Wires & Cables Conduit or Bundles
Silver-Coated Copper	12	23.0
	14	17.0
	16	13.0
	18	10.0
	20	7.5
	22	5.0
Copper Alloy	24	1.8
	26	1.1
	28	0.7
	30	0.4
	32	0.27
<p>These current values are specified for a temperature of 200°C (392°F) maximum allowable conductor temperature and maximum ambient temperature around wires of 155°C (311°F).</p> <p>Current values are for each wire in the conduit or bundle. Copper alloy conductivity = 0.84 x copper conductivity.</p>		

3.5.3 Specific restraints. The following criteria shall govern specific restraints.

3.5.3.1 Interchangeability. All cable and harness assemblies having the same part number shall be directly and completely interchangeable with respect to installation and function.

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3.5.3.2 Conductor size and current capacity. Size 26 AWG wire shall be used throughout the TSS , except where current carrying capacity or excessive voltage drop requires the use of a larger gage. Current carrying capacity shall be in accordance with the requirements of Table I.

3.5.3.3 Circuit segregation. Low level (quiet, sensitive, "clean") high level (noise producing, "dirty") circuits shall be routed in separate bundles, and terminated in separate connectors.

When separate connectors cannot be used, low and high level circuits may share a connector, provided that maximum possible circuit separation at the receptacle is attained through controlled pin assignments.

See applicable circuit data sheets for identification of circuit categories. Connector pin assignments shall also conform to the requirements of 3.5.3.5.

3.5.3.4 Interconnections. The following criteria shall govern interconnections.

3.5.3.4.1 Connectors. Connectors shall conform to the following:

- a. Large, complex or frequently removed components shall be provided with connector junctions.
- b. Circular connectors for subsystem power shall standardize on the 14-19 insert where practicable.
- c. Proven in-flight disconnect connectors shall be provided.

3.5.3.4.2 Splices. Only butt splices shall be used, and their use shall be limited to:

- a. A multiple interconnection (i.e., chassis ground) where a significant saving in wire and connectors can be realized.

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- b. Interconnecting components provided with furnished leads (i. e., transducers).

Circuit integrity and maintainability shall not be impaired by the use of splices. Splices shall preferably be located within the connector potting mass or confined to rigid or semi-flexible portions of the cabling.

3.5.3.5 Connector pin assignment. The following criteria shall govern connector pin assignment.

3.5.3.5.1 Spare or unused pins. Connectors shall have the facility of accepting additional circuitry at a later date. Spare pins may be utilized for isolation of power and return wiring, and also may be used to separate incompatible groups of wiring. If reserved for future assignment only, the pin should be in an accessible insert location. Normally, the keyed side of the circular connector is up during fabrication; therefore, pins on the upper and outer periphery shall be left unused for future applications.

3.5.3.5. 2 Adjacent pins. Where possible, adjacent pins shall be used for bussed, spliced, twisted and shielded functions.

3.5.3.5. 3 Function grouping. Within a given connector, various circuit functions can exist. These functions are defined by type and level of signal or power and categorized into groups. Functions within a group shall be assigned adjacent pins and isolated from incompatible groups by use of spare pins.

3.5.3.5. 4 Socket/pin selection. Where feasible, the pins or sockets of a connector shall be selected so that the power flow is from socket to pin side of the connector. Connector contacts on the TSS half of direct access connectors (unmated during flight) shall be nonprotruding socket types for TSS safety.



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3.5.3.5. 5 Conductors per solder cup. Not more than one conductor shall terminate at any single connector solder cup.

3.5.5.5.6 TSS/Spacecraft in-flight separation electrical interfaces shall be implemented with connectors rigidly mounted on the spacecraft.

Socket contacts shall be employed in the spacecraft connector.

3.5.3.5.7 TSS / support equipment functions. All TSS/ support equipment electrical interfaces shall be implemented with connectors rigidly mounted on the TSS. The contacts in these connectors shall be sockets rather than pins. All ground function connectors required to mate with flight connectors shall be of flight quality.

3.5.3.6 Circuit and cable assignment. The following criteria shall govern circuit and cable assignment:

- a. The assignment of the various circuits to the cables and harnesses, the choice of conductor type (single wire, twisted wires and shielding), and shield ground methods shall be in conformance with the Circuit Data Sheets of JPL Drawing (TBD).
- b. The assignment of circuits to the cables, the choice of conductor type (single wire, twisted wires and shielding) and shield ground methods shall be specified for all circuits.
- c. Spare circuits or conductors shall not be included in harnesses unless specifically required by subsystem design.

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3.5.3.7 High voltages. Cables in which instantaneous voltage differentials exist in excess of 250 volts peak, between two or more conductors, shall conform to JPL Design Requirement DM505139. At ac frequencies higher than 60 Hz, the voltage levels of concern shall be decreased in accordance with JPL Design Requirement DM505139.

3.5.4 Environmental. The wiring harnesses shall be designed to withstand the Earth environment as well as that encountered in space. The design shall satisfy the environmental conditions defined in JPL Specification TS (TBD).

3.5.5 Operational. Mating connector halves shall have keying and insert size and patterns to preclude the inadvertent mismatch to another connector in the vicinity. Where adjacent connectors must be identical, they shall be color coded.

3.5.5.1 Warnings. The applicable drawing shall specifically state any warning or operational procedure that may be required during fabrication installation, test, or operation of the particular cable.

3.5.5.2 Connector mating and demating. Connectors shall be mated and demated only by trained personnel. The number of times flight harness connectors are mated and unmated during the harness fabrication, installation into the electronic assemblies, and subsequently into the TSS, shall be recorded as specified in JPL Specification FS500444.

3.5.5.3 Installation and removal of harnesses. Harnesses shall be designed to withstand the manipulations incidental to installation onto and removal from the electronic assemblies and the TSS.

3.6 RF circuits. Normally, RF circuits are entirely separate from other signal circuits. Where an RF coaxial cable is bundled with other signal circuits, the overall harness shall satisfy all requirements specified herein, and the requirements of the applicable detail drawings.

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3.7 Fabrication Cables, harness assemblies and wiring shall be fabricated in accordance with applicable drawings and JPL Specification FS500444.

3.8 Reliability. Reliability requirements shall be in accordance with JPL Document TSS-3-100. Redundant wiring shall not be provided except in conformance with JPL Document TSS-3-100.

3.9 Pyrotechnic firing circuit cables and harnesses. Pyrotechnic firing circuit cables and harnesses shall be in accordance with the applicable requirements of JPL Functional Requirement TSS-3-260 and the Air Force Eastern Test Range Safety Manual AFETRM 127-1. Details of materials, design and construction shall meet the requirements specified herein.

3.10 Identification. Cables and harness assembly identification shall be in accordance with JPL Document TSS-3-100. The marking methods shall be in accordance with JPL Specification FS500451.

3.11 Configuration. The subsystem wiring and harness shall conform to the individual harness drawings and to the master cable interconnect diagram.

3.12 Electrical test criteria. Each harness and cable assembly shall be capable of passing the following electrical tests:

- a. Insulation and isolation resistance of 100 megohms minimum.
- b. Continuity from pin to pin.
- c. Hi-potential test only when the applicable drawing specifically calls for it.

3.13 Workmanship. Workmanship shall be of the highest quality. Uniformity of shapes, dimensions, and construction shall permit interchangeability of complete units. There shall be no cracks, breaks, chips, bends, burrs, loose parts or other evidence of poor workmanship.

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#### 4. QUALITY ASSURANCE PROVISIONS

4.1 Inspections and tests. The Quality Assurance requirements, inspections and tests for all cable and harness assemblies covered by this document shall be as specified in Section 4 of JPL Specification FS500444 and JPL Document TSS-3-100.

4.2 Rejection and resubmittal. Units that do not meet all the test requirements of this specification shall be rejected. Before resubmittal, complete particulars concerning the previous rejection and the action taken to correct the defects shall be furnished.

#### 5. PREPARATION FOR DELIVERY

5.1 Preparation for delivery shall be in accordance with the contract or procurement document. Shipping and handling equipment (as needed) shall be in conformance to JPL Specification ES500448.

#### 6. NOTES

None.

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